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GEMINI LAUNCH VEHICLE

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Houston, Texas

LAUNCH VEHICLE NO. 6

FLIGHT EVALUATION (U)

U. S. Government Agencies Only

Engineering Report 13227-6

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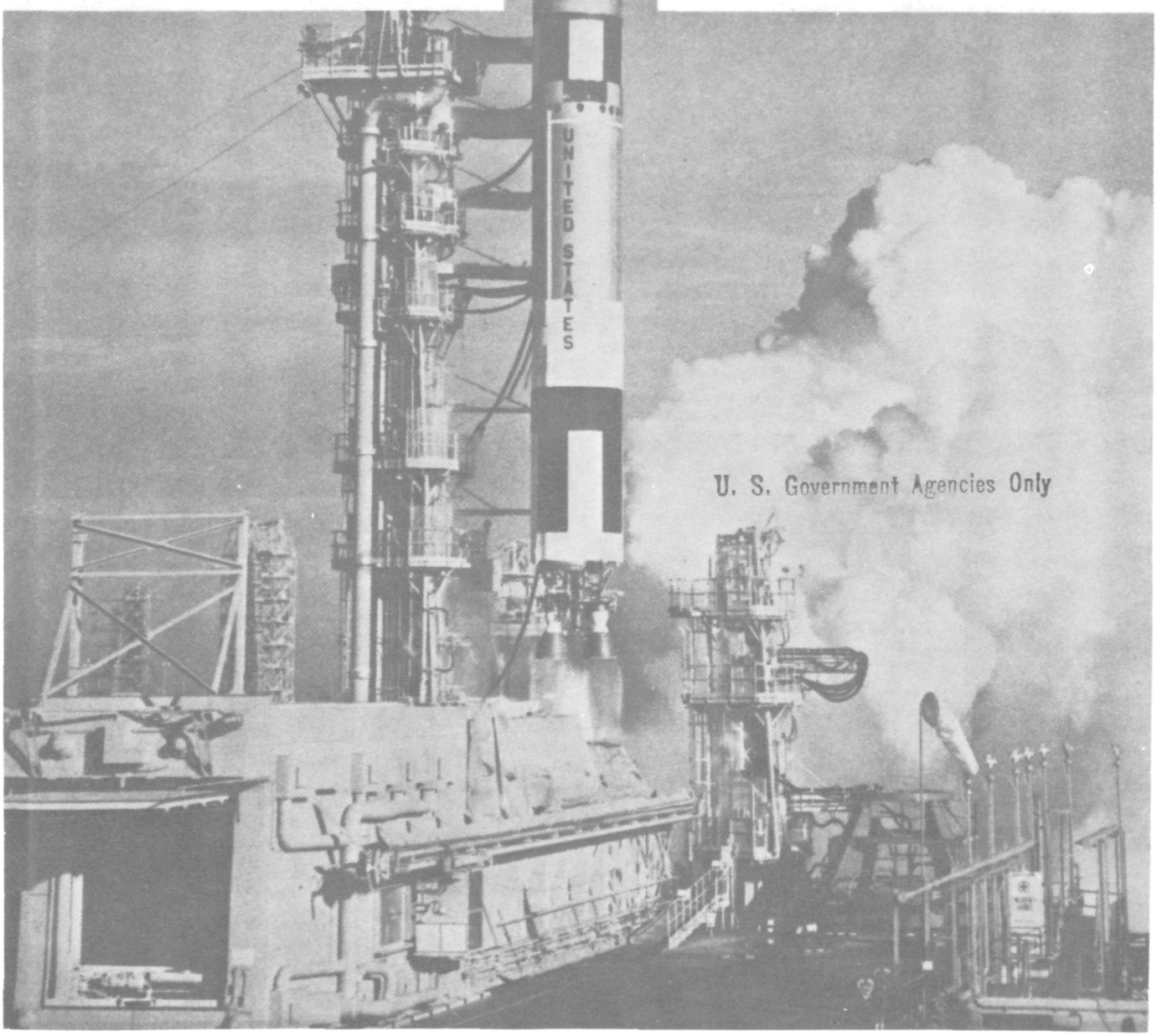

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Prepared by

MARTIN COMPANY, BALTIMORE DIVISION
Baltimore, Maryland 21203

Under CONTRACT AF 04(695)-394
PRIORITY DX-A2

For

SPACE SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND

UNITED STATES AIR FORCE
Los Angeles, California

FOREWORD

This report has been prepared by the Gemini Launch Vehicle Program Test Evaluation Section of the Martin Company, Baltimore Division. It is submitted to the Space Systems Division, Air Force Systems Command, in compliance with Contract AF04(695)-394.

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SUMMARY

The scheduled launch of GT-6A on 12 December 1965 was terminated due to a premature disconnection of tail plug 3D1M. Engine shutdown was automatically initiated 1.24 seconds after MOCS T_0 by holdfire C-4 (programmer reset monitor), when the programmer in the three-axis reference system (TARS), having been initiated by the inadvertent disconnect of 3D1M, started prematurely. Engine shutdown was preceded by a normal start transient in S/A 1, whereas S/A 2 exhibited an abnormal pressure buildup. Investigation of the abnormality led to an inspection of the gas generator, which revealed that a plastic dust cover had been left in the S/A 2 gas generator assembly. All other systems performed properly during the launch attempt.

On 15 December 1965, Gemini-Titan 6A (GT-6A) was launched successfully and on schedule from Complex 19, Cape Kennedy, Florida. Launch vehicle/spacecraft separation was completed 361 seconds after liftoff. Spacecraft re-entry was accomplished after completion of 17.1 orbits.

The 240-minute countdown was picked up at 0529 EST on 15 December and continued without incident through liftoff at 0837 EST. The spacecraft was inserted into an elliptical orbit with a perigee of 87 nautical miles and an apogee of 140.4 nautical miles; all test objectives for the launch were achieved.

Stages I and II engines operated satisfactorily throughout powered flight. Stage I burning time was 160.4 seconds, with shutdown initiated by oxidizer exhaustion. Stage II engine operation was terminated by a guidance command after 181.6 seconds of operation.

The flight control system (FCS) maintained satisfactory vehicle stability during Stages I and II flight. The primary FCS was in command throughout the flight. Vehicle rates during Stage I flight never exceeded 1.9 deg/sec, and the maximum attitude error was 1.7 degrees. The maximum rate and attitude error that occurred during staging did not exceed 3.7 deg/sec and 2.8 degrees, respectively.

Performance of the radio guidance system (RGS) was satisfactory. Pitch and yaw steering signals and SECO discrete commands were properly executed.

IGS pitch, yaw and roll performance for the entire flight appeared normal. The dispersions between IGS and primary system attitude errors remained within acceptable limits during powered flight.

The hydraulic system operated satisfactorily during launch operations and both stages of flight. There were no significant pressure perturbations at liftoff or during flight.

The electrical system functioned as designed throughout the launch countdown and flight. Power transfer to vehicle batteries was smooth.

All channels of the PCM instrumentation system functioned satisfactorily throughout the flight. The landline instrumentation system also functioned satisfactorily prior to and up to liftoff. All airborne instrumentation hold functions monitored in the blockhouse remained within specification throughout the countdown.

The ordnance system umbilical drop weight release, explosive launch nuts and stage separation nuts operated as designed. The prevalves were not replaced after the 12 December 1965 launch attempt, therefore, the valves were open prior to propellant loading. The performances of the command control receivers and the MISTRAM transponder were satisfactory.

Malfunction detection system (MDS) performance during preflight checkout and flight was satisfactory. There were no switchover commands during the flight.

The flight environment encountered by GT-6A was within design requirements. Flight loads were well within the structural capabilities of the launch vehicle. The most critical loading (which occurred at pre-BECO, aft of Station 320) reached 103.6% of design limit load in compression.

The longitudinal oscillation (POGO) on GT-6A reached a maximum value at Station 280 of 0.115 g zero-to-peak at frequencies of 13.7 cps at LO + 146.8 seconds and 16.8 cps at LO + 153.9 seconds. This was the lowest POGO experienced on any Gemini flight to date.

Crew safety monitoring, which was conducted at NASA-MSC, was active during prelaunch and the launch. All guidance monitor parameters were nominal and no corrective action was required during the flight.

The precount operation progressed without problems. All AGE and facilities operated without incident during the countdown. Propellant loading was completed within the scheduled time span and to the specified load and temperature limits.

All electrical umbilicals disconnected in the planned sequence and within 0.854 second. Engine blast and heat damage to the launch stand was minor.

GLV-6 Test Objectives and Results

Objective		Results	
Primary			
P-1	Demonstrate satisfactory boost by the Gemini launch vehicle system of a manned Gemini spacecraft into the prescribed orbital insertion conditions.	P-1	Orbit insertion was within the predicted tolerances for V, h and γ.
P-2	Evaluate launch vehicle subsystem performance during powered flight for mission success and crew safety.	P-2	All systems performed satisfactorily throughout flight. The POGO oscillation (0.115 g zero-to-peak) was the lowest encountered.
P-3	Demonstrate the effectiveness of launch operations including proper operations of necessary ground/range support systems to achieve the prescribed rendezvous mission launch requirements.	P-3	GT-6A was erected and ready for launch countdown seven days after the launch of GT-7 from the same launch pad.
Secondary			
S-1	Evaluate trajectory performance of the launch vehicle system for refining capability and predictions for future missions.	S-1	Vehicle flight was within the 3-sigma predicted trajectory.
S-2	Demonstrate ability to load propellants to weight and temperature limits imposed by payload and vehicle requirements.	S-2	Tanks were loaded within the required tolerances of weight and temperature.

I. INTRODUCTION

This report presents an engineering evaluation of Gemini Launch Vehicle No. 6 (GLV-6) systems performance during the 15 December 1965 countdown, launch and boost phase of the Gemini 6A mission. Discussion of pertinent aspects of the 12 December launch attempt has also been included.

The GT-6A vehicle was launched from Complex 19, Cape Kennedy, Florida at 0837 hours EST on 15 December 1965, 12 days after the launch of GT-7 from the same pad. A successful flight was achieved, and the spacecraft was inserted into an elliptical orbit with a perigee of 87 nautical miles and an apogee of 140.4 nautical miles. Gemini 6A was the seventh mission and the fifth manned flight in the Gemini program, with Astronauts Walter Schirra and Thomas Stafford aboard the Gemini 6 spacecraft. The one-day mission, with its objective of rendezvous with the Gemini 7 spacecraft, was completed successfully on 16 December 1965.

GLV-6 was delivered to Cape Kennedy on 2 August 1965 in preparation for the Gemini 6 mission, a rendezvous and docking exercise with an Agena Target Vehicle (ATV), scheduled to begin on 25 October 1965. On that day, the GLV countdown and the mission were terminated shortly after the Atlas-Agena liftoff when the ATV failed to achieve orbit.

Subsequently, the Gemini 6 mission plan was changed to that of a rendezvous with the Gemini 7 spacecraft and redesignated as the Gemini 6A mission. For the redefined 6A mission, it was required to launch GT-6A from Complex 19 within eight to twelve days following the GT-7 flight. Following the successful GT-7 operation on 4 December 1965, the GT-6A vehicle was erected on Complex 19, and both vehicle and launch complex were readied for a 12 December flight. On this date the countdown proceeded on schedule through T-0 and engine ignition, but an automatic shutdown occurred due to inadvertent release of a tail plug. The GT-6A vehicle was recycled to permit a 15 December launch, which was accomplished on time and without incident and which was followed by the Gemini 6 rendezvous with the Gemini 7 spacecraft.

Significant events and tests accomplished for GLV-6 at ETR appear in Table I-1.

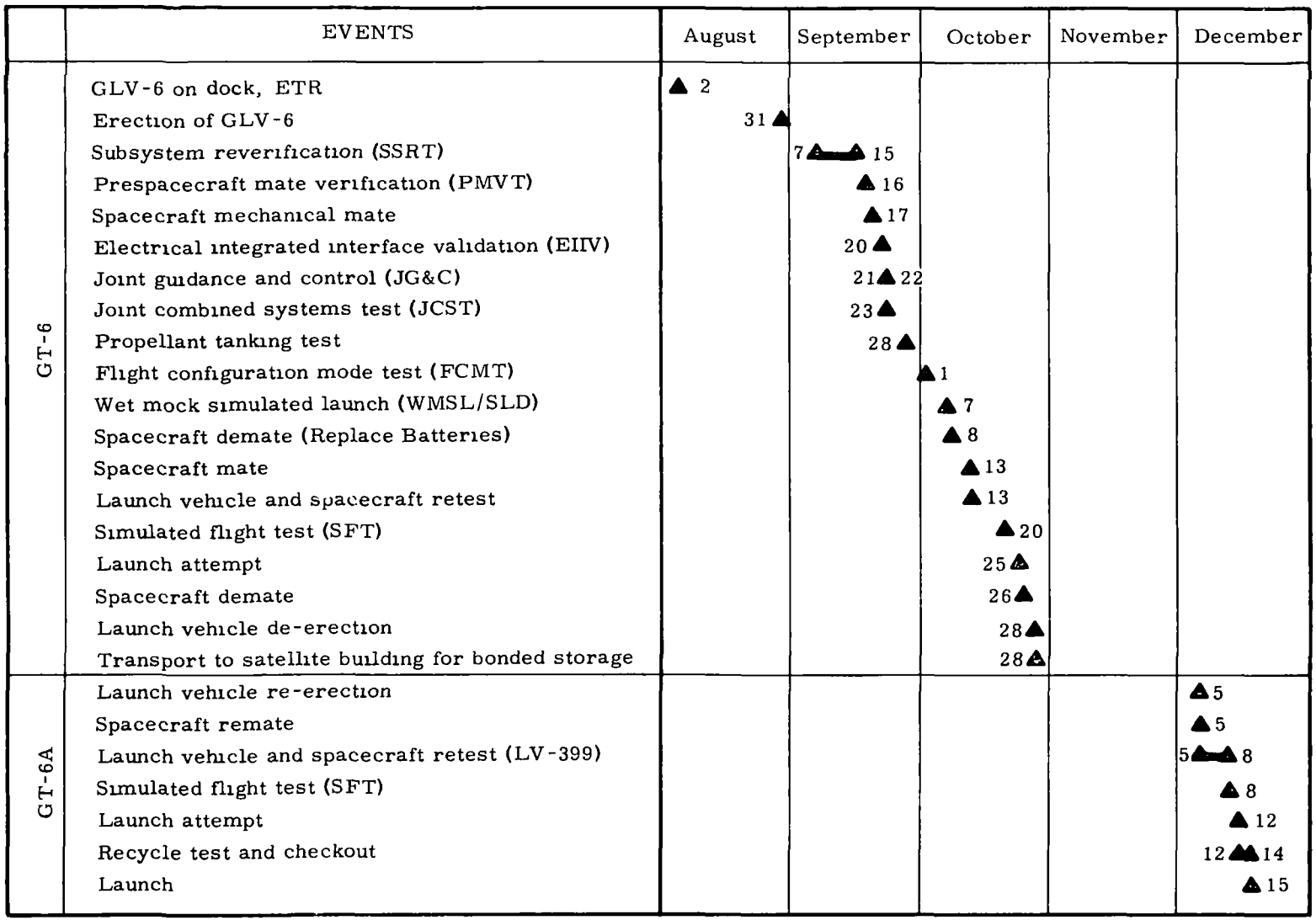

Fig. I-1. Significant Events and Tests at ETR for GT-6 and GT-6A

II. SYSTEM PERFORMANCE

A. TRAJECTORY ANALYSIS

1. Orbit Insertion

Gemini Launch Vehicle No. 6 (GLV-6) performed as predicted and inserted the Gemini 6 spacecraft into earth orbit well within the allowable tolerance limits to permit rendezvous with the GT-7 spacecraft.

GLV-6 was steered in the lateral plane during Stage II flight to a set of ephemeris data referenced to the time of insertion (or targeting). The values of two of these targeting parameters ($\dot{\Omega}$ = -0.18719584 x 10^{-5} rad/sec and T_R^* = 53,741.21875 seconds) are given here for record; there are no observed values of these parameters. The targeted and observed inclination angles were 28.895 and 28.97 degrees, respectively. The targeted wedge angle of 0.2002 degree was exceeded. The observed residual wedge angle was -0.075 degree, which meant that the total wedge angle steered was -0.275 degree.

A comparison of the predicted and observed insertion conditions is given in Table II-1. In this table and in all succeeding references to a predicted (nominal) trajectory, the data were obtained from the GLV-6 45-day prelaunch report (Ref. 10), updated to reflect the actual spacecraft weight (7821 pounds), guidance constants, T-1 hour wind and atmospheric data, and the -1.07% pitch and -1.4% roll programmer biases. The observed trajectory parameters are those derived by the Martin Company from the Final GE Mod III-G 10 pps data. These data have been smoothed and corrected for both refraction errors and systematic biases by the General Electric Corporation before submittal to the Martin Company.

TABLE II-1

Comparison of Insertion Conditions at SECO + 20 Seconds

	Predicted Nominal	GE Mod III-G	Observed Minus Planned	Preliminary Tolerance
Altitude (naut mi)	87.128	87.260	+0.132	±0.346
Inertial velocity (fps)	25,731	25,728	-3	±30.3
Inertial flight path angle (deg)	0.002	-0.054	-0.056	±0.125

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TABLE II-2

Predicted Dispersions at BECO and at SECO + 20 Seconds

Performance Parameter	Predicted Nominal Value* (Ref. 10)	Predicted Dispersions		
		Vehicle Dispersion (Ref. 11)	RGS Dispersion (Ref. 12)	RSS of RGS and Vehicle Dispersions
BECO				
Time from liftoff (sec)	156.578	±4.44	N/A	±4.44
Geocentric radius (ft)	21,115,242	±13,734		±13,734
Inertial velocity (fps)	10,004	±196		±196
Crossrange (ft)	-3468	±15,804		±15,804
Crossrange velocity (fps)	-55	±394		±394
Ground range (naut mi)	52.64	±3.40		±3.40
Inertial flight path angle (deg)	18.14	±2.71	N/A	±2.71
SECO + 20 seconds				
Time from liftoff (sec)	358.894	±7.45	±0.081	±7.45
Geocentric radius (ft)	21,438,718	±1554	±1410	±2100
Inertial velocity (fps)	25,731	±18.4	±24.09	±30.3
Crossrange (ft)	-162,376	±73,592	±498	±73,597
Crossrange velocity (fps)	-1976	±23.08	±9.90	±24.9
Ground range (naut mi)	535.9	±15.36	±0.1416	±15.36
Inertial flight path angle (deg)	0.0022	±0.0540	±0.1125	±0.125

N/A: Not applicable.

*Updated to reflect actual spacecraft weight (7821 pounds), guidance constants, T-1 hour wind and atmospheric data, and -1.07% pitch and -1.4% roll programmer biases.

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2. Derivation of Trajectory Uncertainties

The expected maximum vehicle dispersions and RGS dispersions at BECO and at SECO + 20 seconds were obtained from Refs. 11 and 12, respectively. A root sum square (RSS) of these dispersions is termed the preliminary tolerance. After determination of the preliminary tolerance, the total tolerance may be computed by the arithmetic addition of the preliminary tolerance to the 3-sigma data error of the instrumentation source being considered. Thus,

$$\text{Preliminary tolerance} = \sqrt{(\text{vehicle dispersions})^2 + (\text{RGS dispersions})^2}$$

Total tolerance = preliminary tolerance + 3-sigma data error.

The resulting preliminary tolerance is shown in Table II-2. Because the actual insertion conditions were within the preliminary tolerance, the data error estimates are not needed and, therefore, have been excluded from this report.

3. Flight Plan

The primary objective for GLV-6A was to place the Gemini 6A spacecraft into an elliptical earth orbit with an 87-nautical mile perigee* and 146-nautical mile apogee.* Having achieved orbital insertion at 25,730 fps,** the spacecraft then separates from Stage II (adding 10 fps to spacecraft velocity in the process) and coasts to the desired apogee. The following flight plan was employed to attain these desired conditions.

A vertical rise is planned for the first 23.04 seconds following liftoff, during which time a programmed roll rate of 1.25 deg/sec is initiated to roll the vehicle from a pad orientation of 84.903 degrees to the flight azimuth of 81.4 degrees.

At this time, an open-loop pitch program is begun (via a three-step rate command) which terminates at 162.56 seconds. The nominal commanded pitch rates and their times of application are shown in Table II-3.

Guidance commands from the radio guidance system (RGS) are initiated at LO + 168.35 seconds and continue until two seconds prior to SECO, however, velocity cutoff computations continue to SECO. Between SECO and SECO + 20 seconds, the engine shutdown impulse continues to add velocity to the vehicle (approximately 85 fps), and the spacecraft is separated from the sustainer just after SECO + 20 seconds.

---

*Relative to Complex 19.

**Does not include the separation velocity imparted by the spacecraft.

TABLE II-3

Planned GLV Pitch Program

Program	Rate (deg/sec)	Time from Liftoff (sec)
Step 1	-0.709	23.04 to 88.32
Step 2	-0.516	88.32 to 119.04
Step 3	-0.235	119.04 to 162.56

A comparison of the planned and actual sequences of events is contained in Table II-4, and a profile of the GT-6A flight superimposed on the range planning map appears in Fig. II-1.

4. Trajectory Results

Analysis of the range data and Mod III radar data indicates that the performance of GLV-6 was normal and the vehicle flew close to the prescribed ascent trajectory throughout Stages I and II. The only significant deviations in the trajectory occurred in the first stage, in which, at BECO, the vehicle was 4900 feet low and 6100 feet to the left of the nominal trajectory.

Reconstruction of the BECO condition (Table II-5) shows that a satisfactory simulation may be obtained by consideration of the actual engine data, weather conditions, propellant loading, inert weight, engine misalignments, wind and guidance errors. This table is comprised of those items which can be measured and those which can only be estimated due to lack of suitable instrumentation. The primary factors contributing to the pitch and yaw plane trajectory dispersions at BECO are listed and the effect of each is summarized. The pitch plane reconstruction matches the flight data quite well, the yaw plane reconstruction does not match the flight data quite as well as for previous Gemini flights.

A Y_F coordinate displacement at BECO results in an approximate amplification of four on the Y_F coordinate displacement at SECO + 20 seconds. The differences shown in the apparent and measured increments of Table II-5 are well within the allowable tolerance limits presented in Table II-2.

Table II-6 presents the trajectory parameters computed from the GE Mod III-G, MISTRAM 1 and 2, and C-band radar data. At BECO all of the data sources yielded comparable results. MISTRAM yielded somewhat different values for the insertion parameters than did GE Mod III-G and C-band radar. MISTRAM values of inertial velocity,

TABLE II-4

GT-6A Flight Events Summary

Measurement	Event	GMT (hr-min-sec)	Time from Liftoff (sec)	
			Actual	Planned
0800/0801	Power transfer	1335 57.6	-88 9	-89.
FC B-10	MOCS T-0	1337 23.13	-3 34	-3.46
2104	$87FS_1$ (T-0)	23.192	-3.279	-3.37
0356	Stage I S/A-1 MDTCPS make	24.092	-2 379	-2 27
0357	Stage I S/A-2 MDTCPS make	24 112	-2 359	-2 27
2101	TCPS S/A-1 and S/A-2	24.172	-2 299	-2 20
0169	Launch nuts	26.28	-0.19	-0 20
4421	First motion	26.362	-0.109	-0.10
4422	Shutdown lockout (backup)	26.378	-0.093	-0 10
4423	Liftoff	26 471	0	0
0734	Start roll program	44.13	17.66	17 68
0734	End roll program	46.93	20.46	20 48
0732	Start pitch program No. 1	·49.46	22.99	23.04
0732	Stop pitch program No. 1	1338 54.67	88.20	88.32
0732	Start pitch program No. 2	54.67	88 20	88 32
0728	FCS gain change No. 1	1339 16.28	109.81	110.00
0732	Stop pitch program No. 2	·25.30	118.83	119 04
0732	Start pitch program No. 3	25.30	118.83	119 04
0735	Staging enable (TARS discrete)	50.88	144.41	144 64
0741	IPS staging arm timer	51 44	144.97	145.00
0356	Stage I S/A-1 MDTCPS break	1340 03 586	157.115	156.52
0357	Stage I S/A-2 MDTCPS break	03 588	157.117	136.52
0032	$87FS_2/91FS_1$ (BECO)	03.633	157.162	156.58
0502	Start PC_3 rise	04.300	157.829	157.23
0169	Stage separation	04.315	157 844	157.31
0855	Stage II MDFJPS make	04.370	157 899	157.48
0732	Stop pitch program No. 3	08.17	161 70	162 56
0740	RGS enable	08 11	161 64	162 56
0755/0756	First guidance command	14 68	168 21	168.35
0739	Stage II shutdown enable	1342 42.75	316.28	317 44
0777	Guidance SECO	1343 05.188	338 717	336 87
0519	$91FS_2$	05 208	338.737	336.89
0522	Shutdown valve relay	05.234	338 763	336.91
0521	Shutdown squib	05.224	338 753	336.91
0799	ASCO	05 24	338 77	336 92
0855	Stage II MDFJPS break	05.348	338 877	337.19
AB-03	Spacecraft separation	27.4	360.9	356.89

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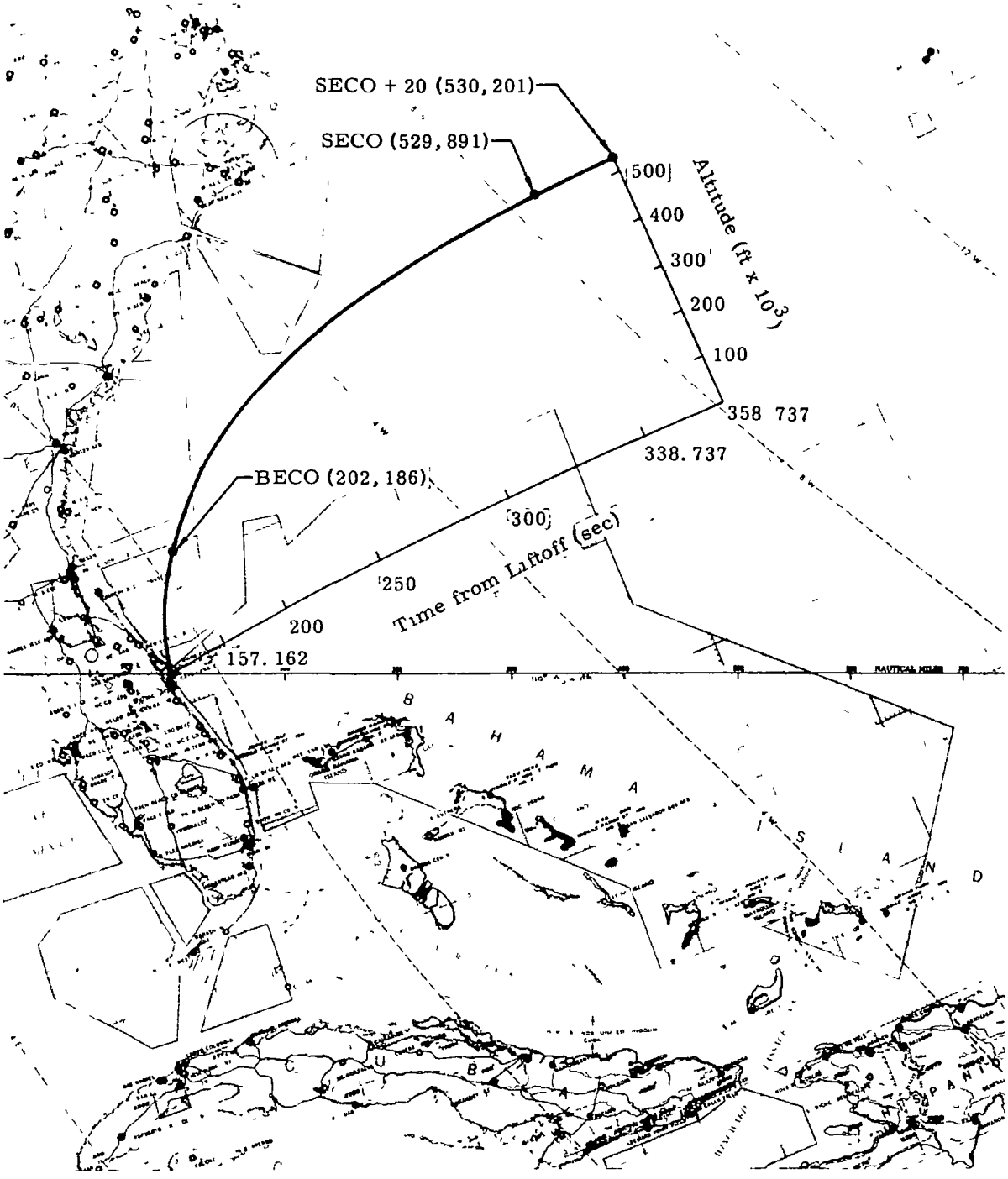

Fig. II-1. GT-6A Boost Flight Path Profile

TABLE II-5

GT-6A BECO Conditions

	Δ Time (sec)	Δ Altitude (ft)	Δ Velocity (fps)	Δγ (deg)	ΔY_F (ft)	$\Delta \dot{Y}_F$ (fps)
Measured Parameters						
Thrust Stage I (-0.43%)	+0 57	-1200	--	-0.218	-14	-0.3
Outage (+381 lb)	-0.22	-653	-37	+0.019	+8	+0.3
1239 GMT wind and weather data	--	-1635	+55	-0.190	-1761	+2
Inert weight (-51 lb)	--	-120	-4	-0.006	--	--
Trend Indications						
TARS pitch programmer (-0 14%)	--	+96	-2	+0 07	--	--
TARS roll programmer (-0 14%)	--	--	--	--	-146	-3 8
TARS pitch gyro drift (+2.56 deg/hr)	--	-359	+3	-0 078	-17	-0.35
TARS yaw gyro drift (-4 16 deg/hr)		-22	-1	-0 003	+720	+18 8
TARS roll gyro drift (+6.0 deg/hr/g)	--	--	--	--	-285	-7.2
Engine misalignment pitch (+0 13 deg)	--	-1870	+19	-0.468	-285	-5.1
Engine misalignment yaw (+0.05 deg)	--	-64	--	-0 012	+990	+28 8
Engine misalignment roll (-0.139 deg)	--	-10	-3	+0 005	+3650	+95
Specific impulse (+0 15%)	+0 230	+323	+17	+0 04	-7	--
Apparent increments	0 58	-4509	+47	-0 84	2853	128
Measured increments	0.58	-4904	+31	-0.38	6100	173

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TABLE II-6

Comparison of GT-6 Planned and Observed Performance at BECO and at SECO + 20 Seconds

	Predicted Nominal Trajectory (Ref 10)*	Tracking Facility					
		GE Mod III-G (10 pps)	GE Mod III-G (2 pps)	MISTRAM I and II	C-Band Radar	BET	Bermuda
BECO							
Time from LO (sec)	156 578	157 162	157 162	157 162	157 162	157 162	N/A
Inertial velocity (fps)	10,004	9992	9980	9981	10,008	9980	
Altitude	205,499	202,186	202,178	202,206	202,247	202,198	
Inertial flight path angle (deg)	18 14	17 94	17 94	17 94	17 92	17 95	
Ground range (naut mi)	52 64	52 62	52 62	52 62	52 60	52 64	
Geocentric radius (ft)	21,115,242	21,111,906	21,111,900	21,111,919	21,111,966	21,111,918	
Downrange position, X_F (ft)	323,005	322,782	322,769	322,774	322,688	322,772	
Cross-range position, Y_F (ft)	-3468	4392	4392	4931	4871	4937	
Vertical position, Z_F (ft)	203,018	199,689	199,681	199,703	199,626	199,701	
Downrange velocity, X_F (fps)	8210	8211	8200	8201	8228	8199	
Cross-range velocity, Y_F (fps)	-55	138	138	152 17	116	153	
Vertical velocity, Z_F (fps)	2993	2956	2953	2953	2958	2952	
Yaw steering velocity, V_Y (fps)	--	-649	-648	-662	-630	-663	
Biased yaw steering velocity, V_Y (fps)	--	--	--	--	--	--	
SECO + 20 seconds							
Time from LO (sec)	356 894	358 737	358 737	358 737	358 736	358 736	358 736
Inertial velocity (fps)	25,731	25,728	25,726	25,717	25,748	25,717	25,718 4
Altitude (ft)	529,397	530,201	530,144	530,070	529,017	529,960	N/A
Inertial flight path angle (deg)	0 0022	-0 054	-0 0315	+0 0352	-0 1071	+0 0250	+0 048
Ground range (naut mi)	535 9	538 6	538 1	538 6	538 7	538 5	N/A
Geocentric radius (ft)	21,438,718	21,439,442	21,439,384	21,439,293	21,438,237	21,439,183	21,438,931
Downrange position, X_F (ft)	3,322,338	3,339,793	3,337,000	3,340,038	3,340,878	3,340,062	N/A
Cross-range position Y_F (ft)	-162,376	-134,390	-134,176	-128,578	-127,689	-128,571	
Vertical position, Z_F (ft)	270,157	268,406	268,787	268,271	267,068	268,156	
Downrange velocity, X_F (fps)	23,987	23,978	23,978	23,978	23,995	23,978	
Cross-range velocity, Y_F (fps)	-1976	-1976	-1974	-1929	-1970	-1929	
Vertical velocity, Z_F (fps)	-3772	-3814	-3800	-3773	-3841 5	-3777	
Yaw steering velocity, V_Y (fps)	-3 7	-5 3	-7	-51 06	-11 8	-51 4	
Biased yaw steering velocity, V_Y (fps)	-15 7	--	--	--	--	--	

*Updated to reflect actual spacecraft weight (7821 pounds), guidance constants, T-1 hour wind and atmospheric data, and -1 07% pitch and -1 4% roll programmer biases

N/A--Not available
Apogee 140 39 naut mi
Perigee 86 89 naut mi

inertial flight path angle, and geocentric radius agreed very closely with values derived from the Bermuda tracking data. BET data (which are derived from MISTRAM data) are presented in Table II-6

Table II-6 presents a comparison of the GE Mod III-G 10 pps and the 2 pps data. Starting with GT-8, only the GE Mod III-G 2 pps data will be available. Comparing the two sets of data on Table II-6, it is concluded that the GE Mod III-G 2 pps data are satisfactory.

The actual, as well as the predicted, nominal trajectory is presented in graphical form in Figs. II-2 through II-24. On these graphs, the nominal trajectory is that documented in Ref. 10, updated to reflect the actual spacecraft weight (7821 pounds), guidance constants, T-1 hour wind and atmospheric data, and the -1.07% pitch and -1.4% roll programmer biases. The observed flight data were obtained from the Mod III-G 10 pps data, smoothed and corrected for refraction errors and systematic biases.

A list of the primary tracking sources with the trajectory time interval covered by each is contained in Table II-7.

5. Geodetic and Weather Parameters

Significant geodetic and weather parameters are shown in Table II-8. The atmospheric pressure and temperature variation with altitude is depicted in Fig. II-25. The pressure was essentially standard, while the temperature was slightly warmer than standard. Figure II-26 presents the altitude history of the magnitude and direction of the wind. At low altitudes the winds were light, increasing to a peak of 86 knots at 42,500 feet. The wind was essentially a tail wind with a small component from the left of the trajectory.

6. Look Angles

An initial decoder pitch-down command of about 0.10 deg/sec, lasting approximately 0.5 second, was issued at LO + 168.21 seconds. Following this, a 0.56 deg/sec pitch-down command was issued for approximately 0.3 second. Thereafter, the pitch steering command decreased to approximately 0.25 deg/sec within 2 seconds.

During this period, the maneuver resulted in decreasing angles of attack as shown in Fig. II-20. The maximum look angle in pitch (LAP) occurred at LO + 335 seconds, when it attained a value of 22.2 degrees. This maximum value was within the boundary existing at that time, as shown in Fig. II-27. The corresponding look angle in yaw (LAY) was also within the established limitation (±20 degrees), as shown in Fig. II-28. The maximum value of LAY was 6.3 degrees which occurred 160 seconds after liftoff.

TABLE II-7

Data Available for Trajectory Analysis

Source	Type	Station	Flight Coverage (sec from range-0)
AFETR	MISTRAM position, velocity and acceleration	Valkaria I Eleuthera II	+65 to 383 +65 to 383
	FPQ-6 radar position, velocity and acceleration	MILA 19.18 GBI 3.18 Grand Turk 7.18 Patrick 0.18	+13 to +380 +147.6 to +379.1 +329.1 to 491.1 90 to 379
	FPS-16 radar position, velocity and acceleration	3.16	+147.6 to +379.1
	BET	(composite)	+66 to +379
GE	Mod III-G radar position, velocity	Cape Kennedy	LO to +381
NASA-MSC	Spacecraft IGS aspect parameters		LO to +367.9

TABLE II-8

Geographic and Weather Conditions at Launch

Location	
Site	Complex 19
Site coordinates·	
Latitude (deg) Longitude (deg)	28.507 N 80.554 W
Pad orientation (deg)	84.903 true azimuth
Weather	
Ambient pressure (psi)	14.71
Ambient temperature (°F)	68
Dew point (°F)	67
Relative humidity (%)	97
Surface wind·	
Speed (fps)	7
Direction (deg)	200
Winds aloft (max):	
Altitude (ft)	42,500
Speed (fps)	147
Direction (deg)	292 true azimuth
Cloud cover	0.2 alto-cumulus
Reference Coordinate System	
Type	Martin reference coordinate system
Origin	Center of launch ring, Complex 19
Positive X-axis	Downrange along flight azimuth tangent to ellipsoid
Positive Y-axis	To left of flight azimuth tangent to ellipsoid and ⊥ X-axis
Positive Z-axis	Forms a right-handed orthogonal system
Reference ellipsoid	Fischer
Launch	
Initial flight azimuth (deg)	81.4 true azimuth
Roll program (deg)	3.5 cw

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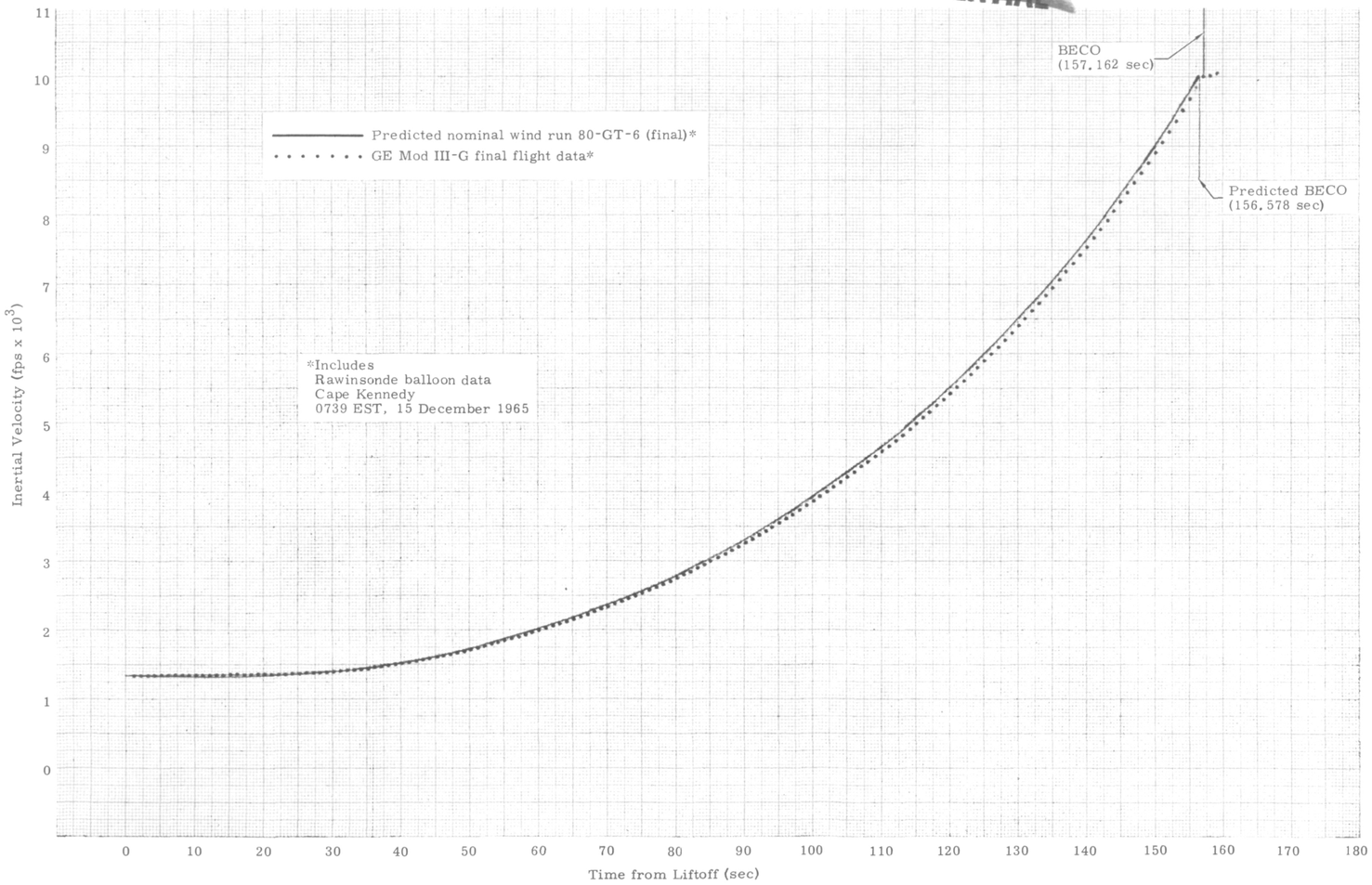

Fig. II-2. Inertial Velocity Versus Time: Stage I Flight

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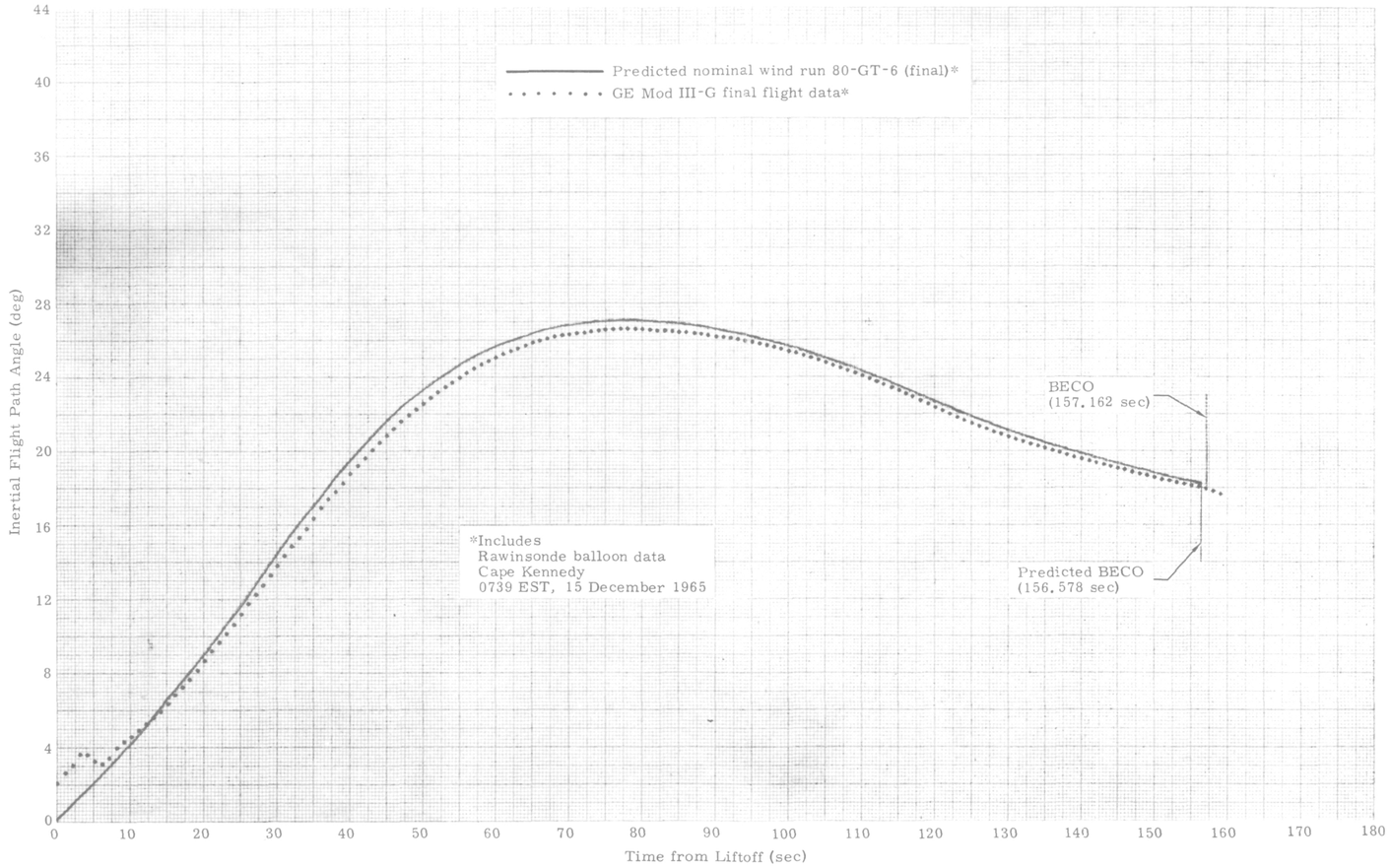

Fig. II-3. Inertial Flight Path Angle Versus Time: Stage I Flight

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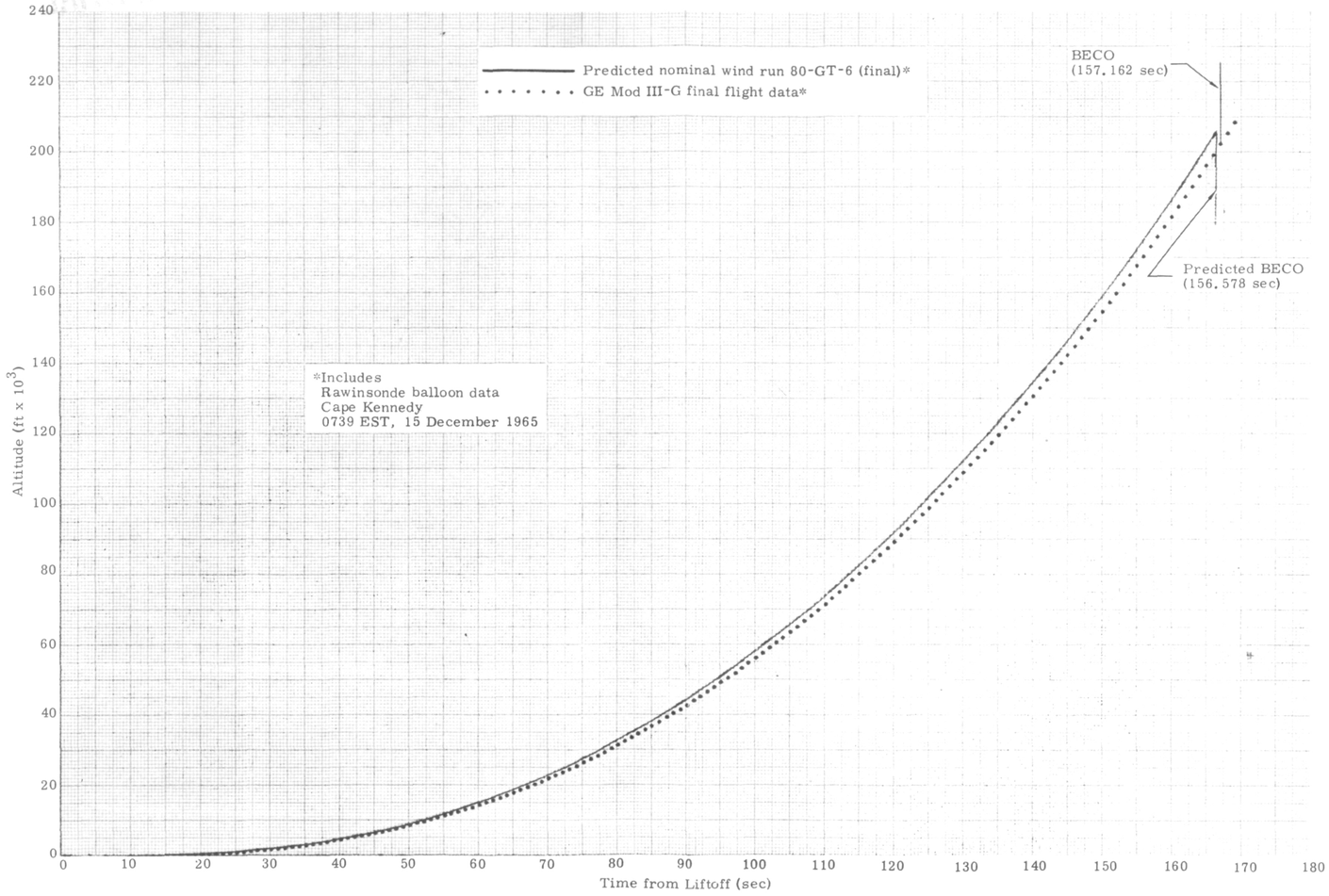

Fig. II-4. Altitude (h) Versus Time: Stage I Flight

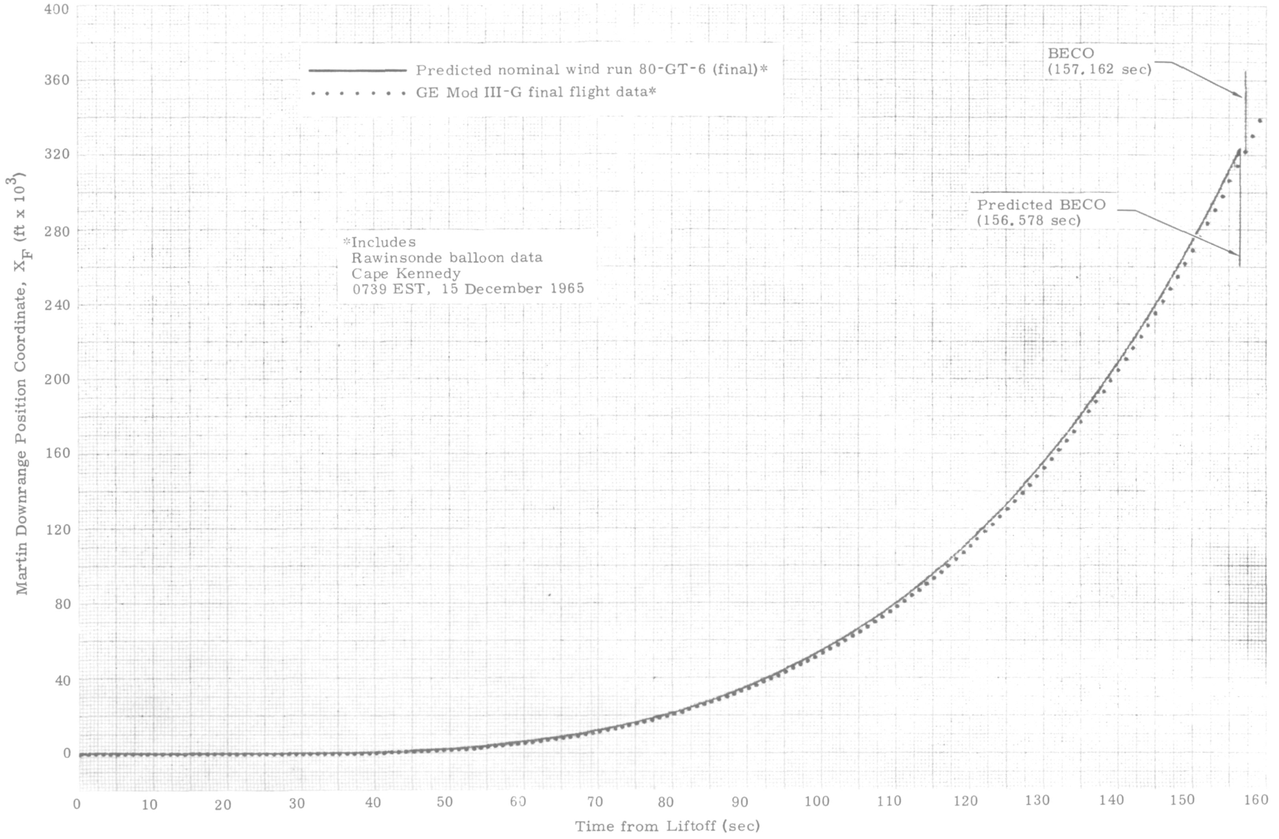

Fig. II-5. Downrange Position Coordinate (X_D) Versus Time: Stage I Flight

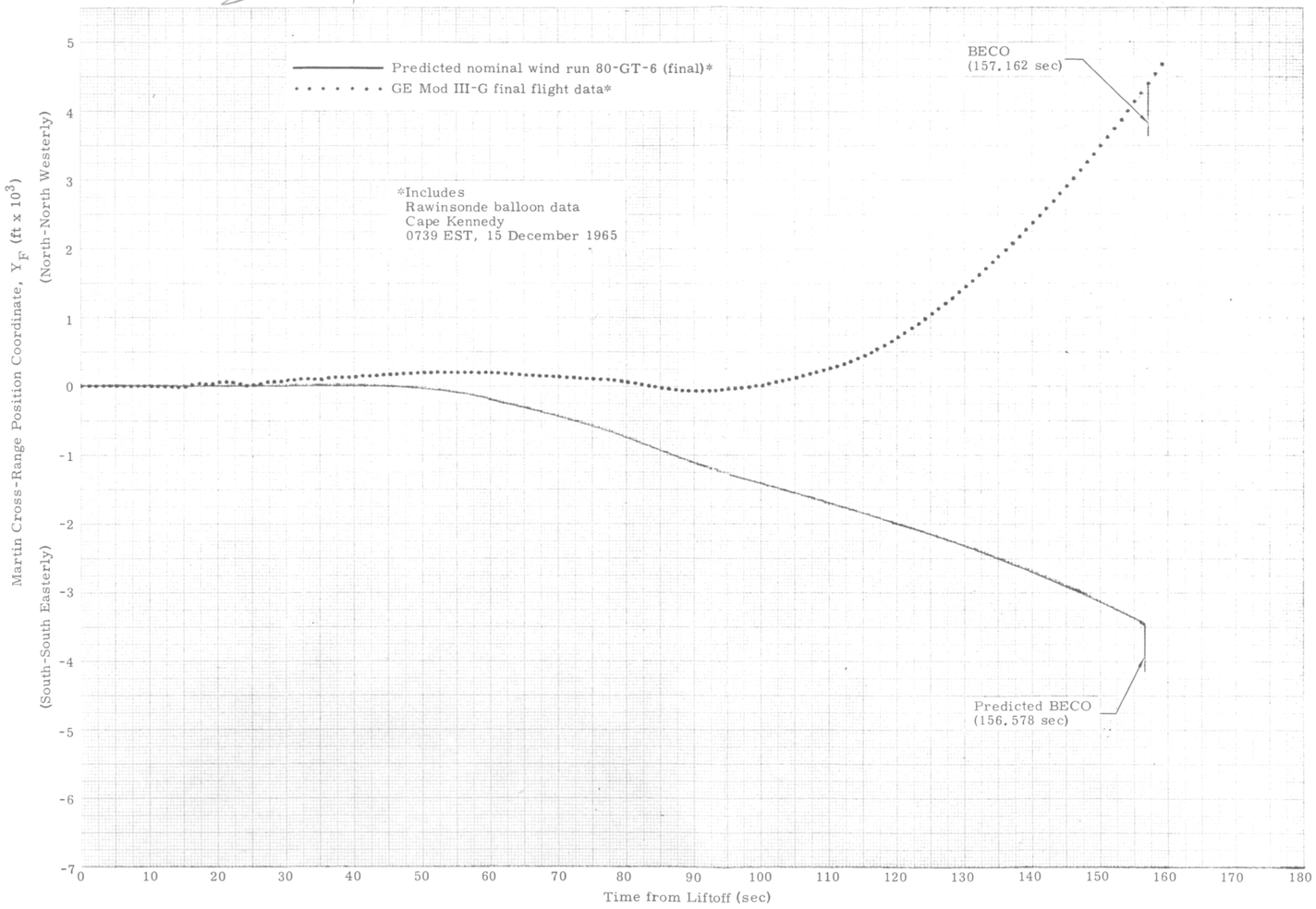

Fig. II-6. Cross-Range Position Coordinate (Y_p) Versus Time: Stage I Flight

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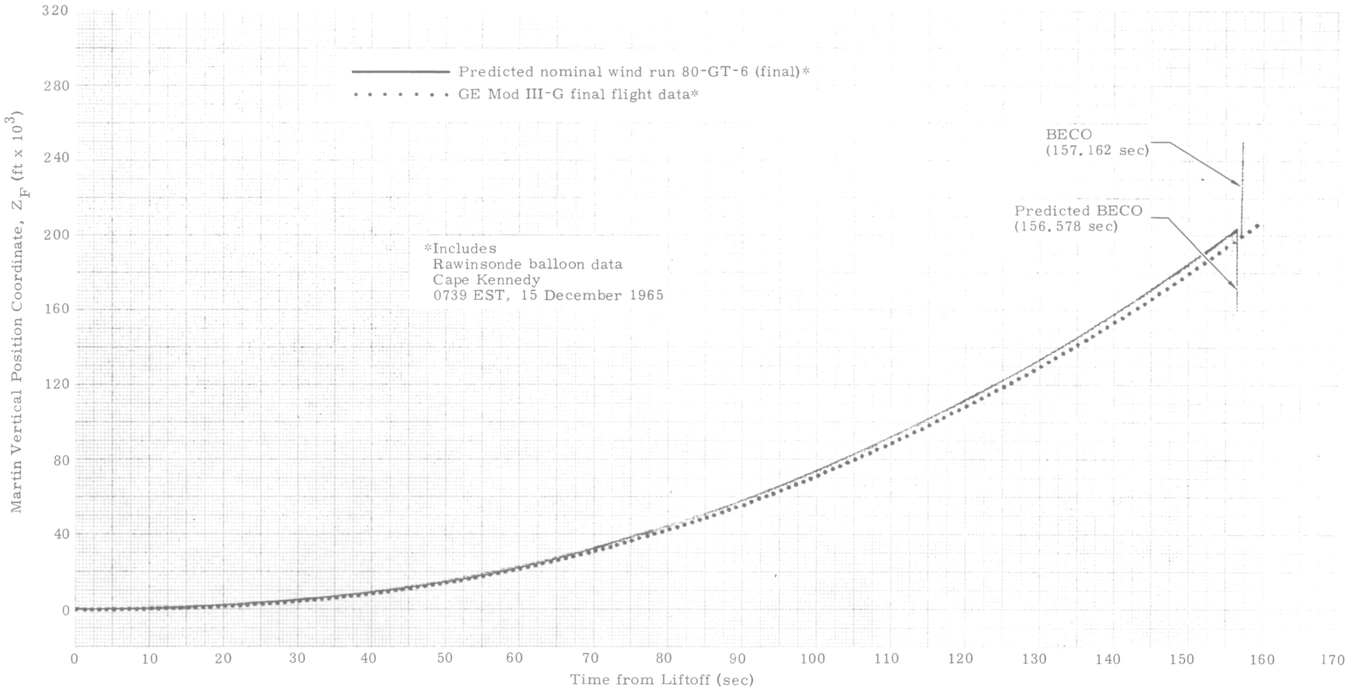

Fig. II-7. Vertical Position Coordinate (Z_P) Versus Time: Stage I Flight

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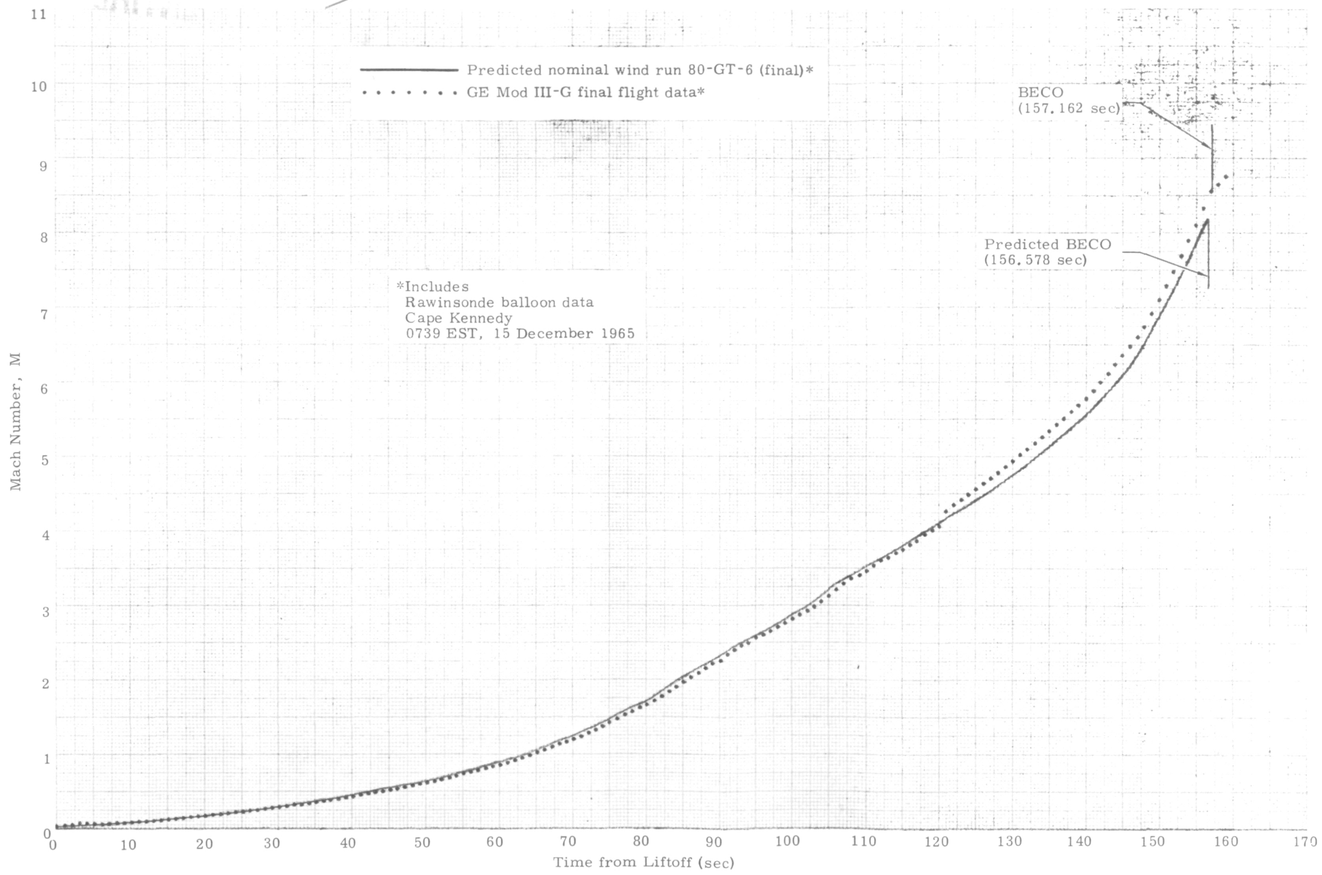

Fig. II-8. Mach Number (M) Versus Time: Stage I Flight

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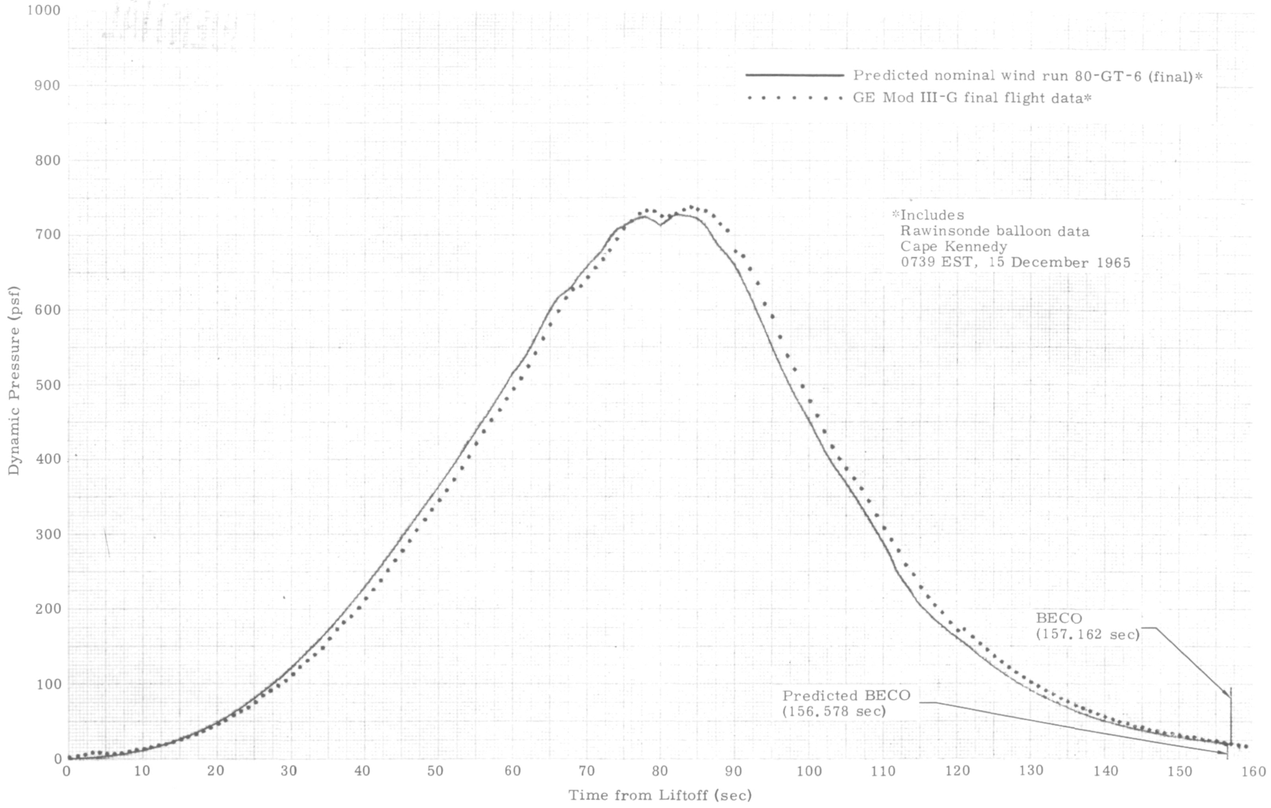

Fig. II-9. Dynamic Pressure (q) Versus Time: Stage I Flight

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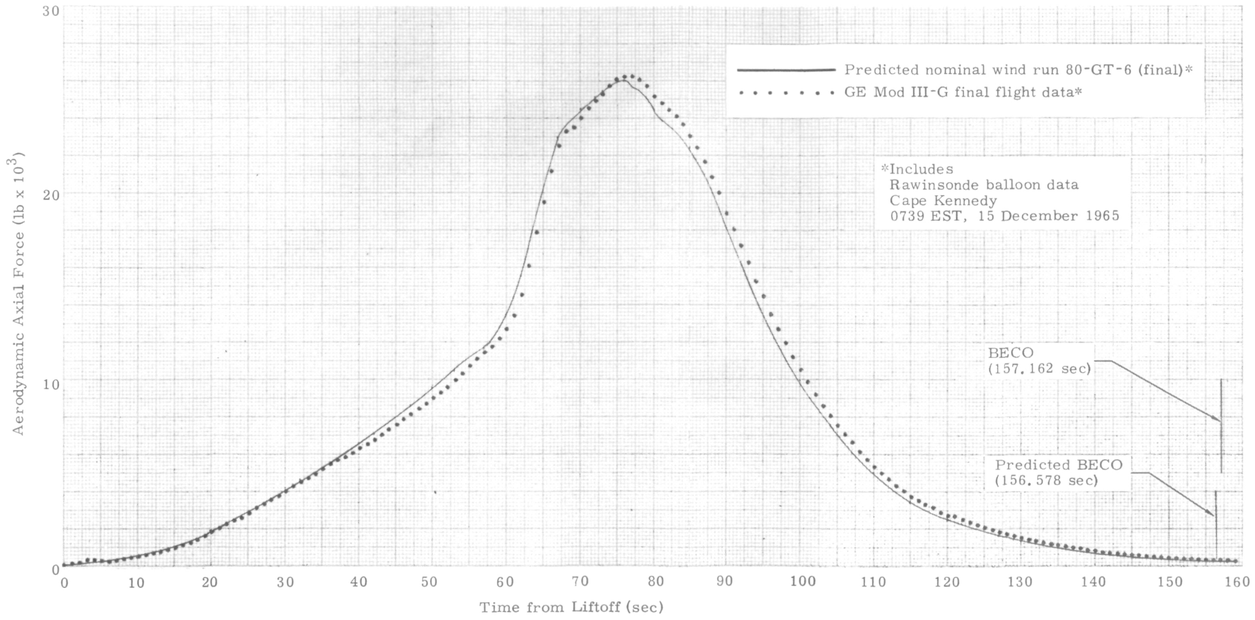

Fig. II-10. Axial Force Versus Time: Stage I Flight

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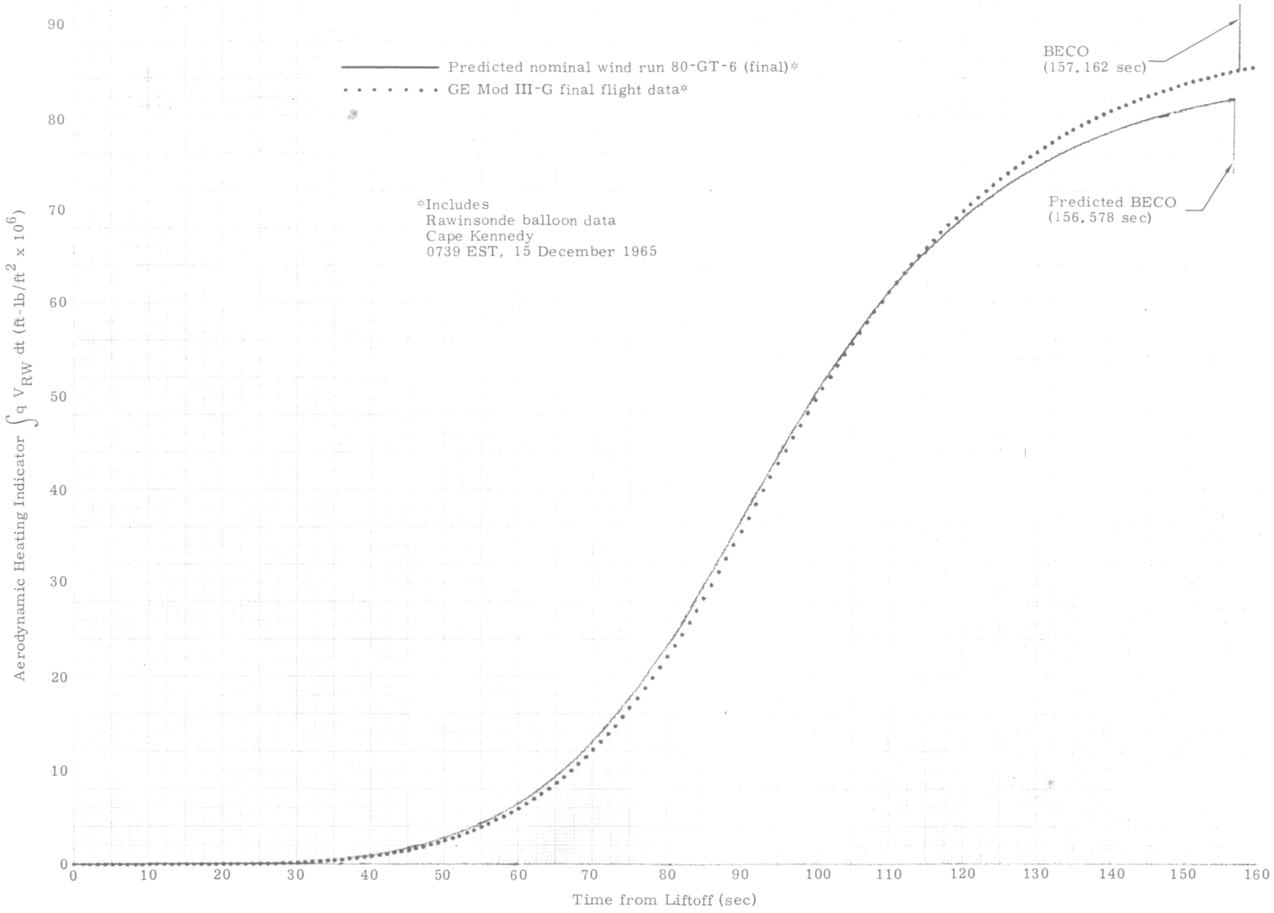

Fig. II-11. Aerodynamic Heating Indicator Versus Time: Stage I Flight

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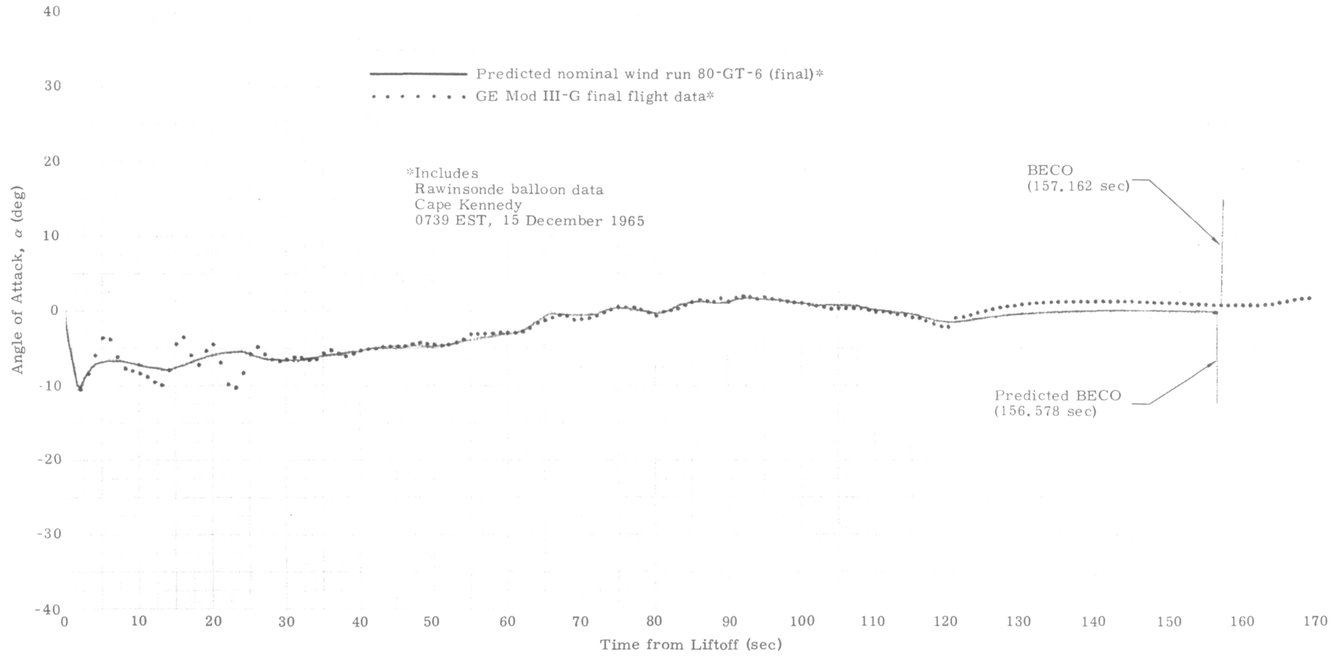

Fig. II-12. Stage I Angle of Attack History

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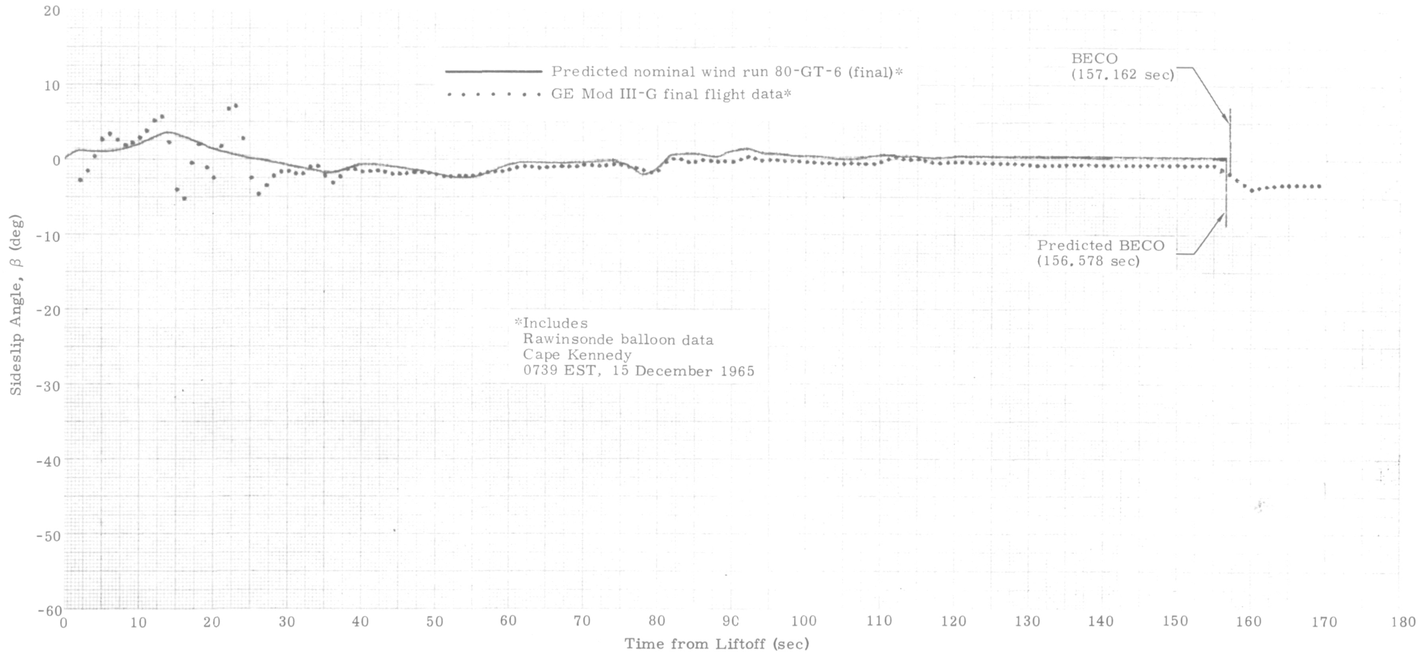

Fig. II-13. Stage I Angle of Sideslip History

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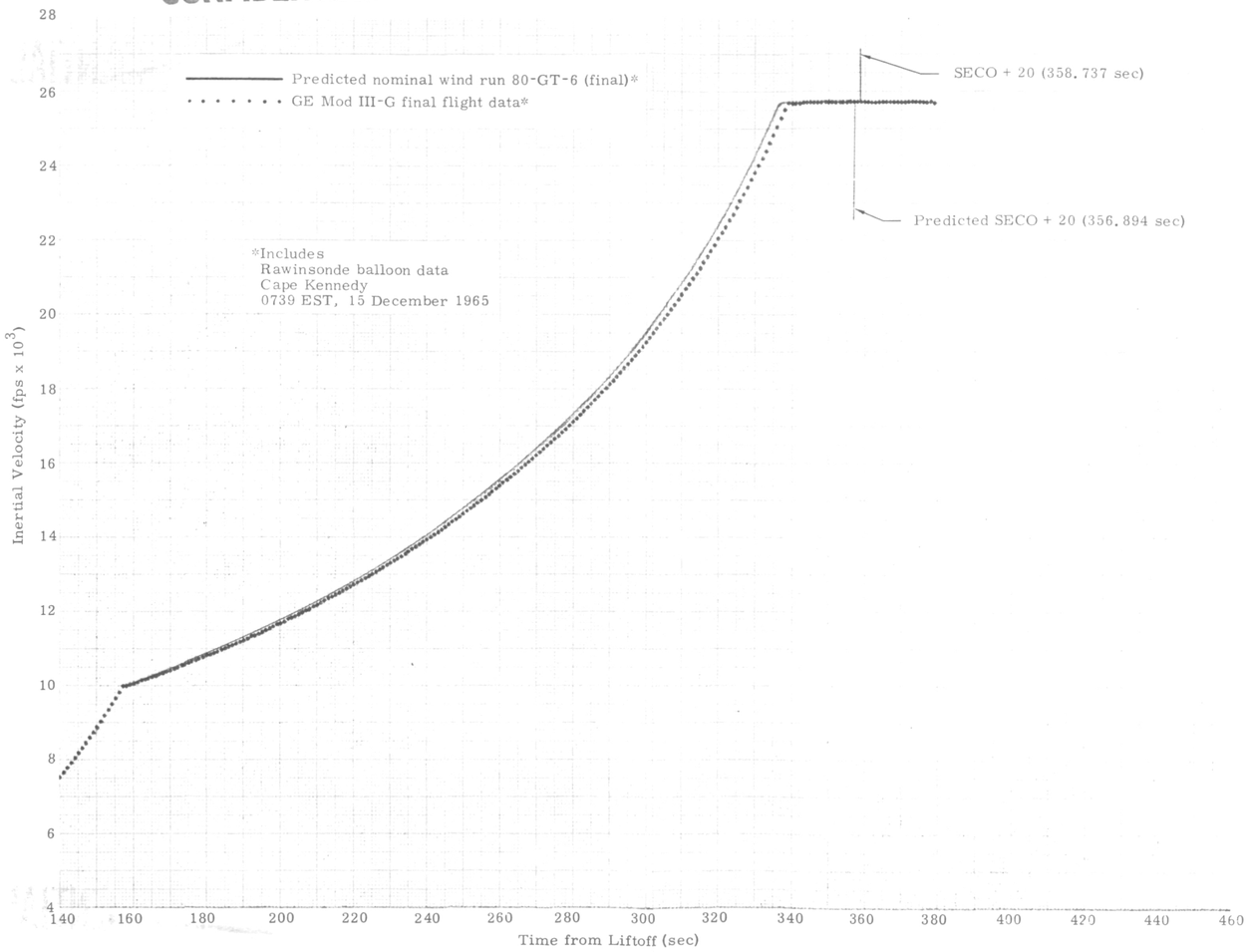

Fig. II-14. Resultant Inertial Velocity (V_I) Versus Time: Stage II Flight

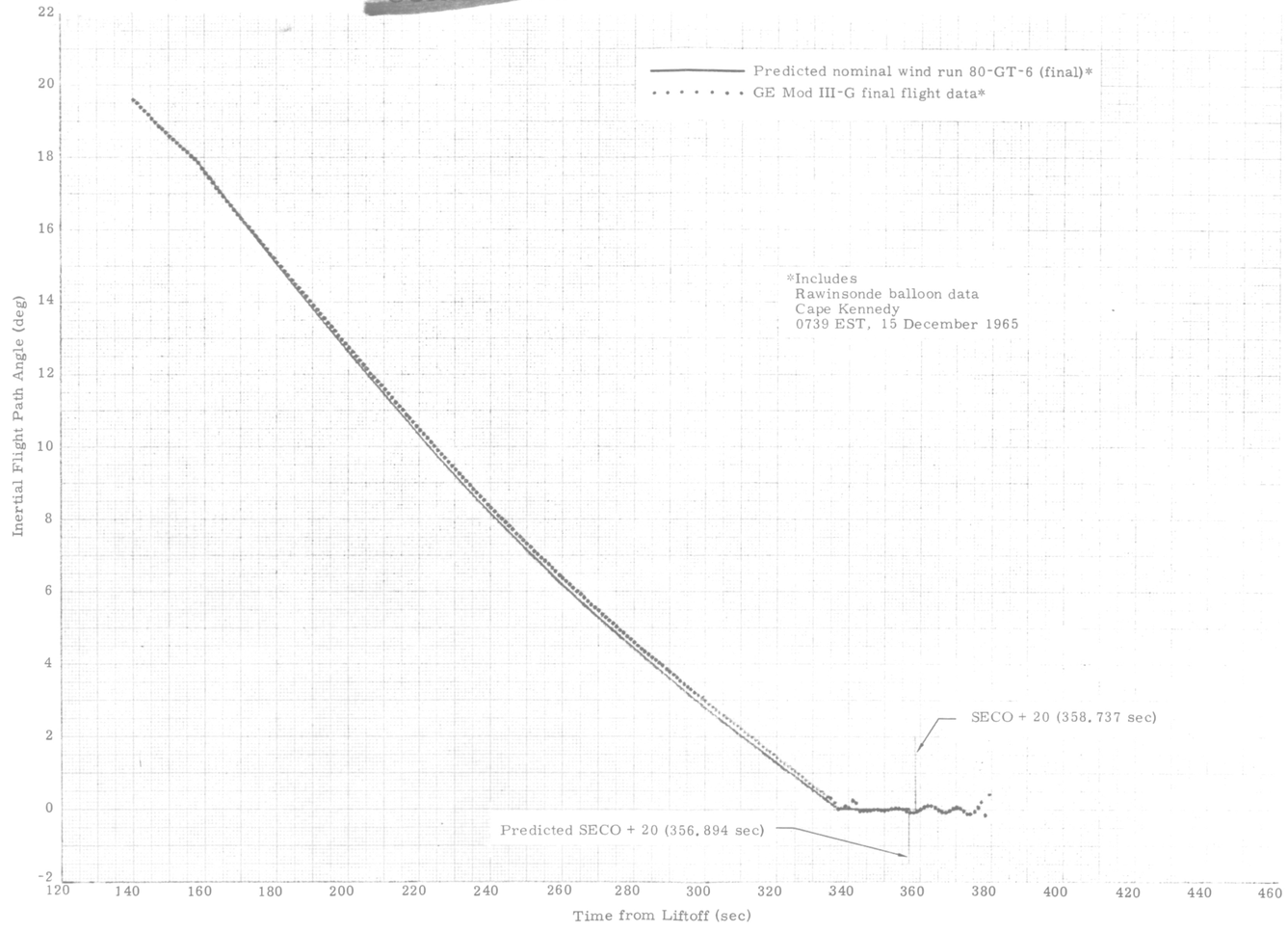

Fig. II-15. Inertial Flight Path Angle (γ_I) Versus Time: Stage II Flight

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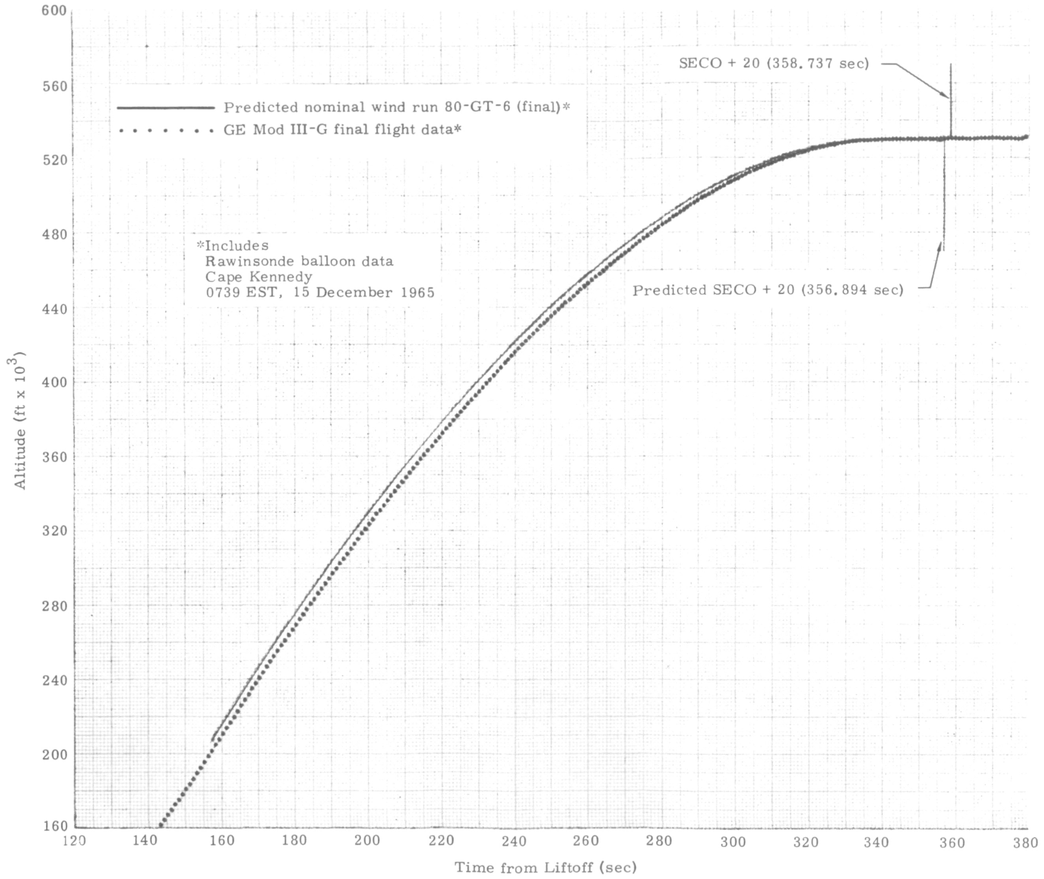

Fig. II-16. Altitude Versus Time: Stage II Flight

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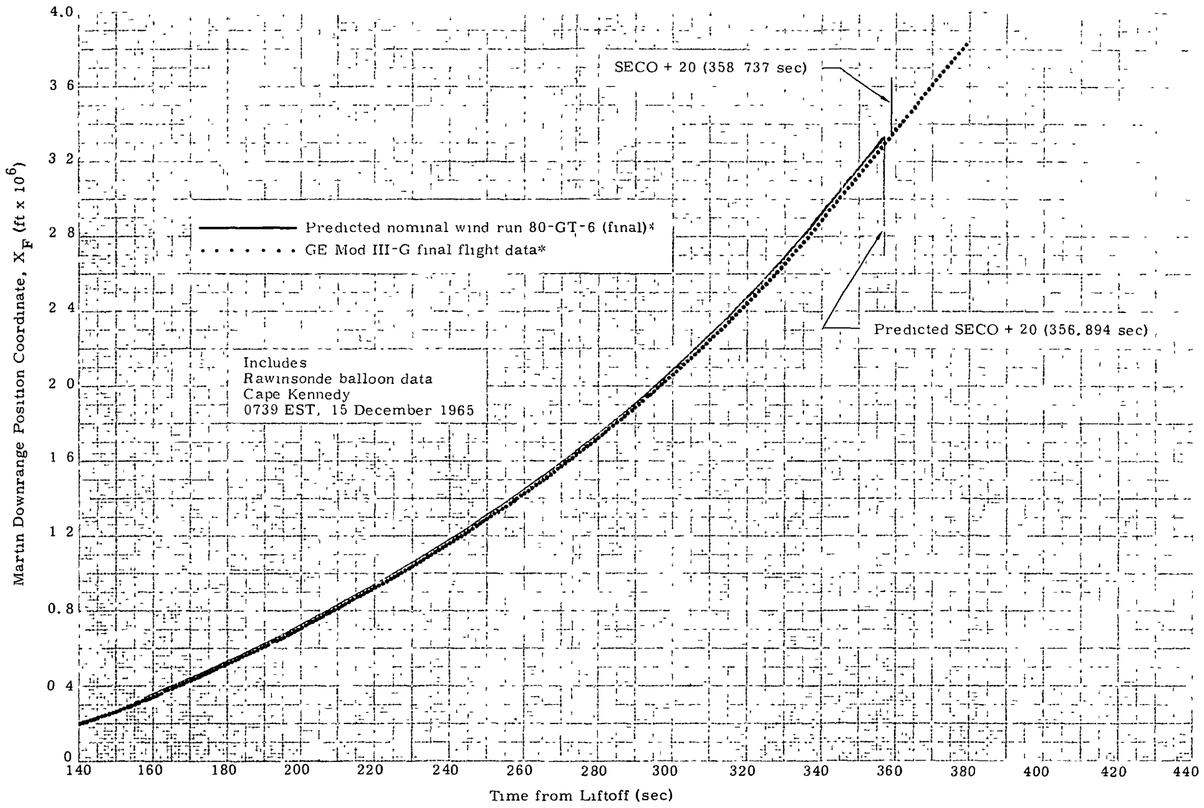

Fig. II-17. Downrange Position Coordinate (X_P) Versus Time: Stage II Flight

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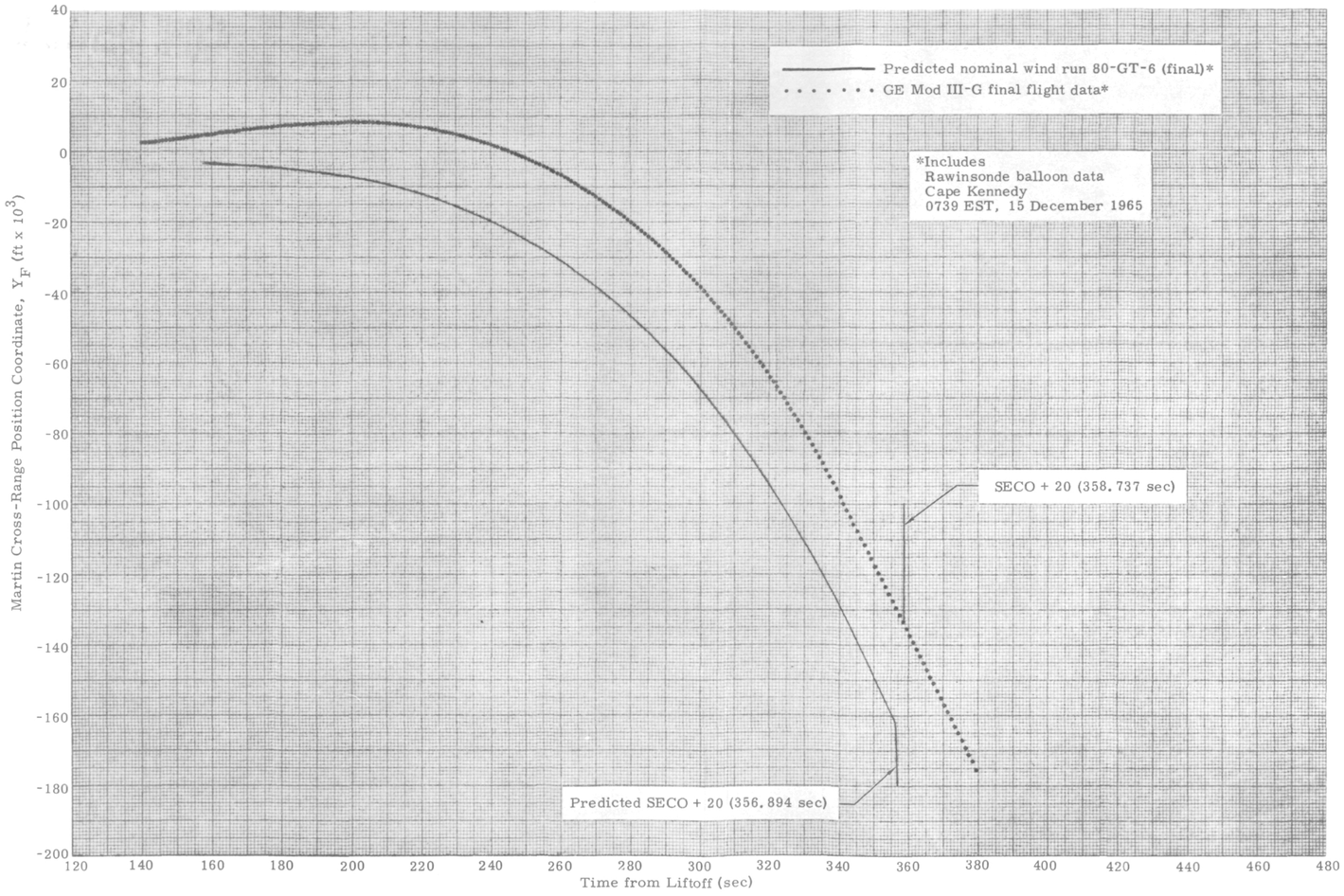

Fig. II-18. Cross-Range Position Coordinate (Y_F) Versus Time: Stage II Flight

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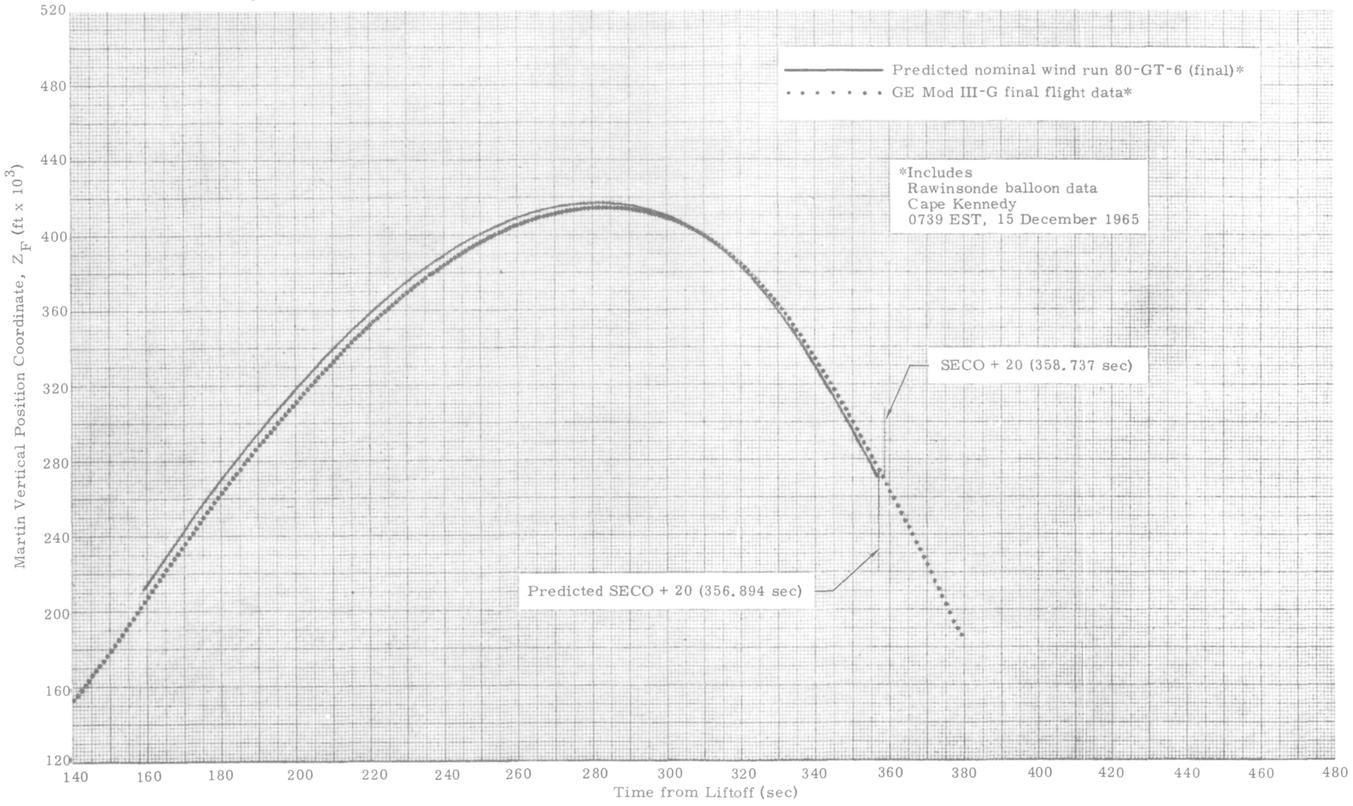

Fig. II-19. Vertical Position Coordinate (Z_F) Versus Time: Stage II Flight

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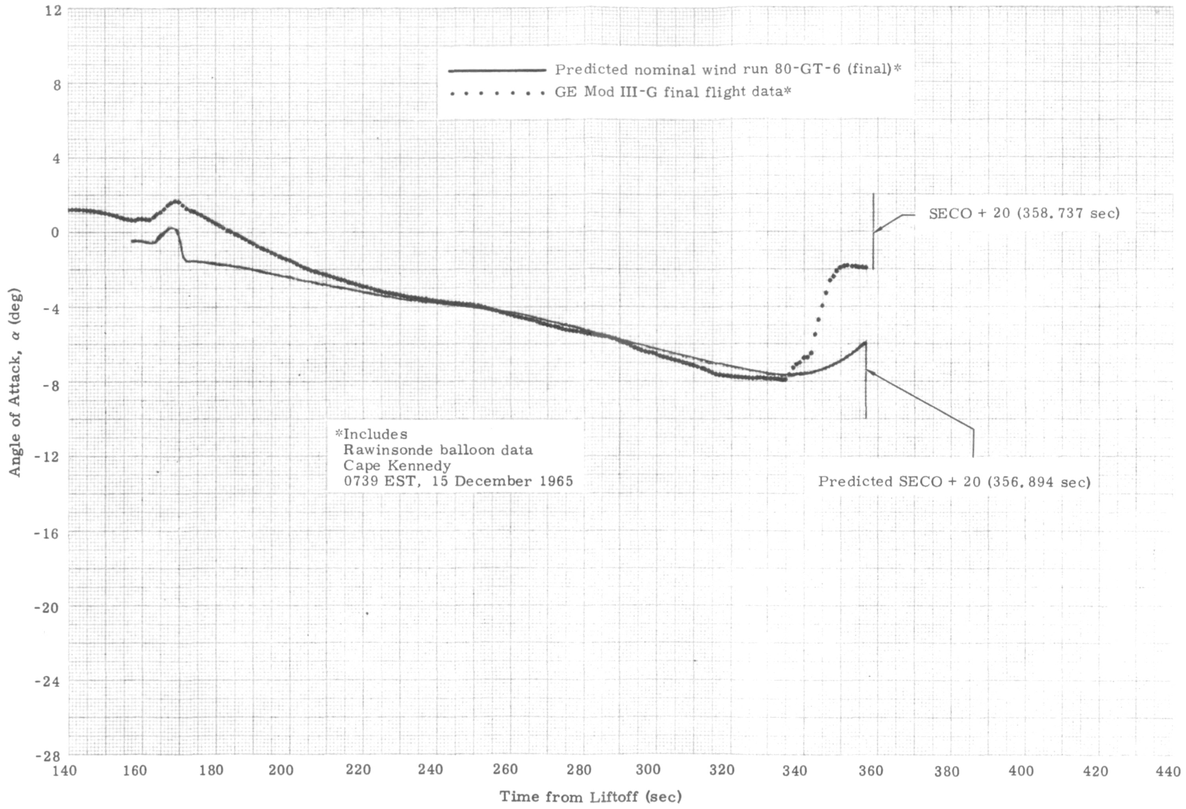

Fig. II-20. Stage II Angle of Attack History

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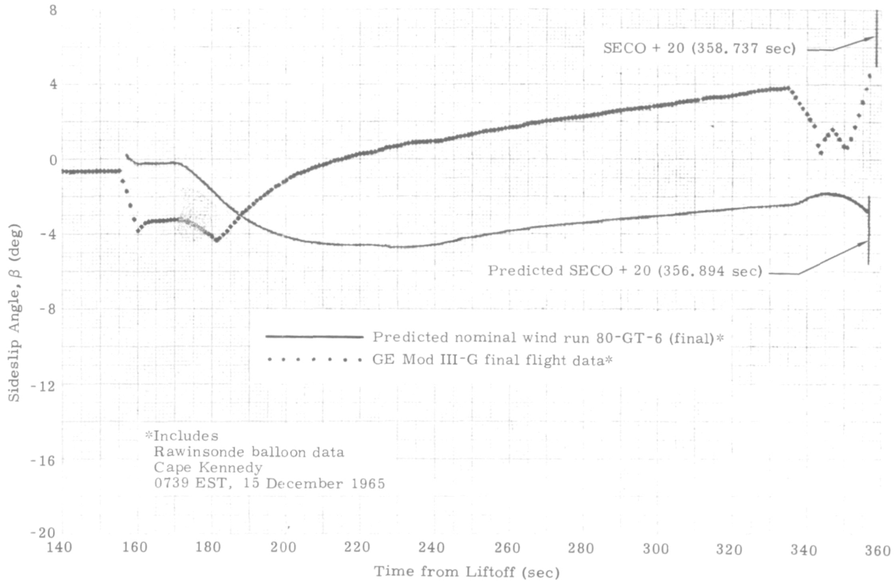

Fig. II-21. Angle of Sideslip Versus Time: Stage II Flight

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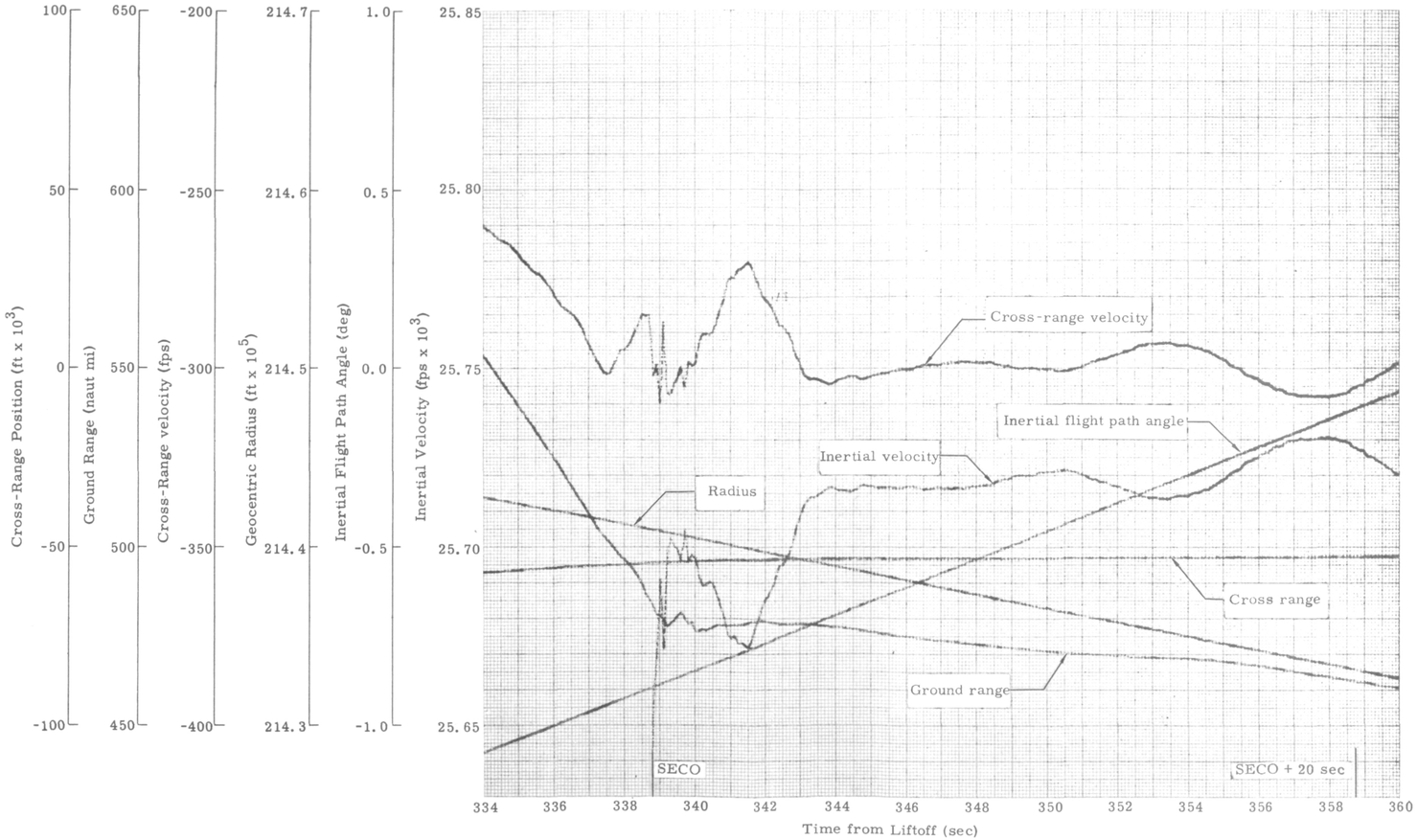

Fig. II-22. GE Mod III-G Flight Data from SECO to SECO + 20 Seconds

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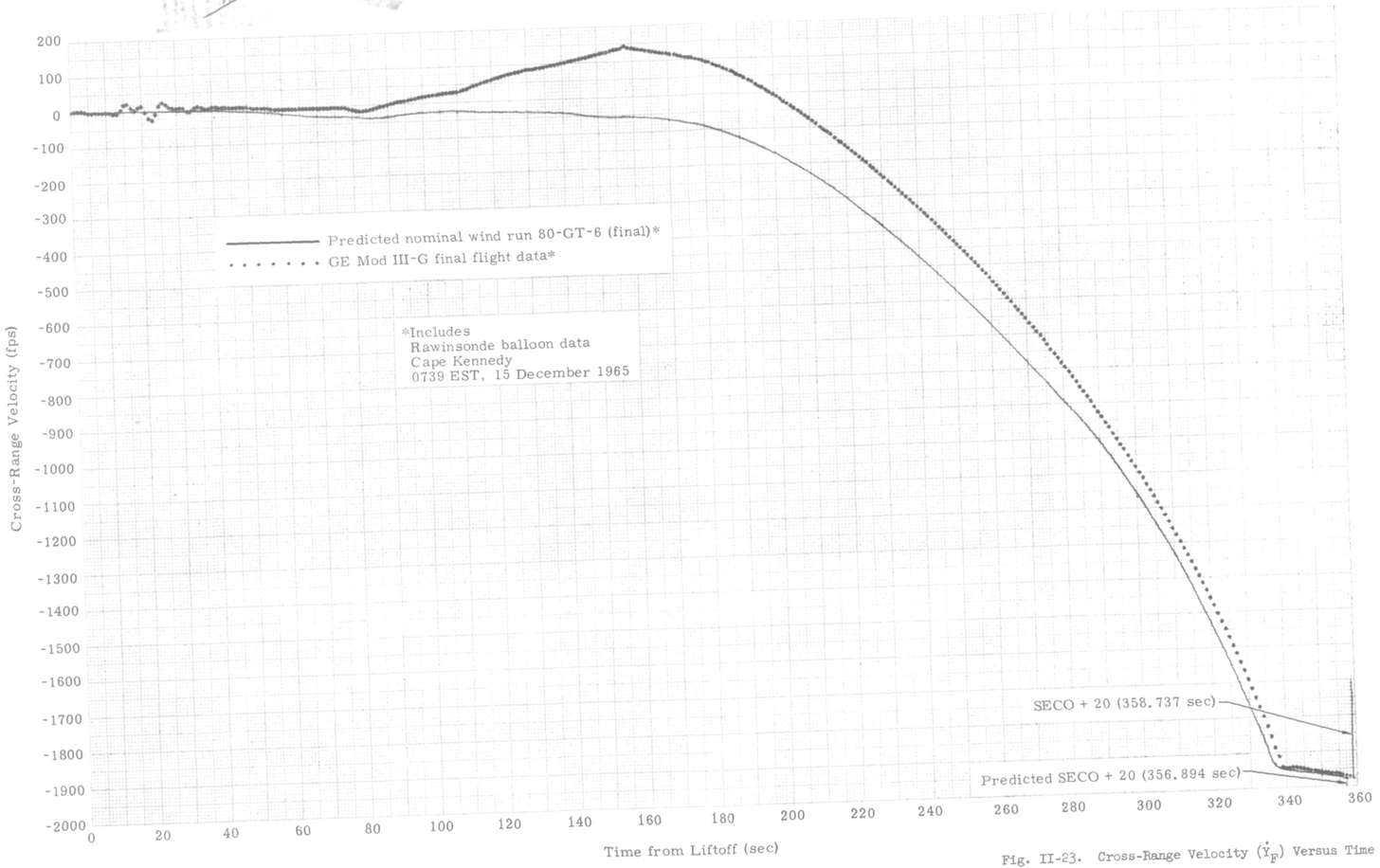

Fig. II-23. Cross-Range Velocity ($\dot{Y}_F$) Versus Time

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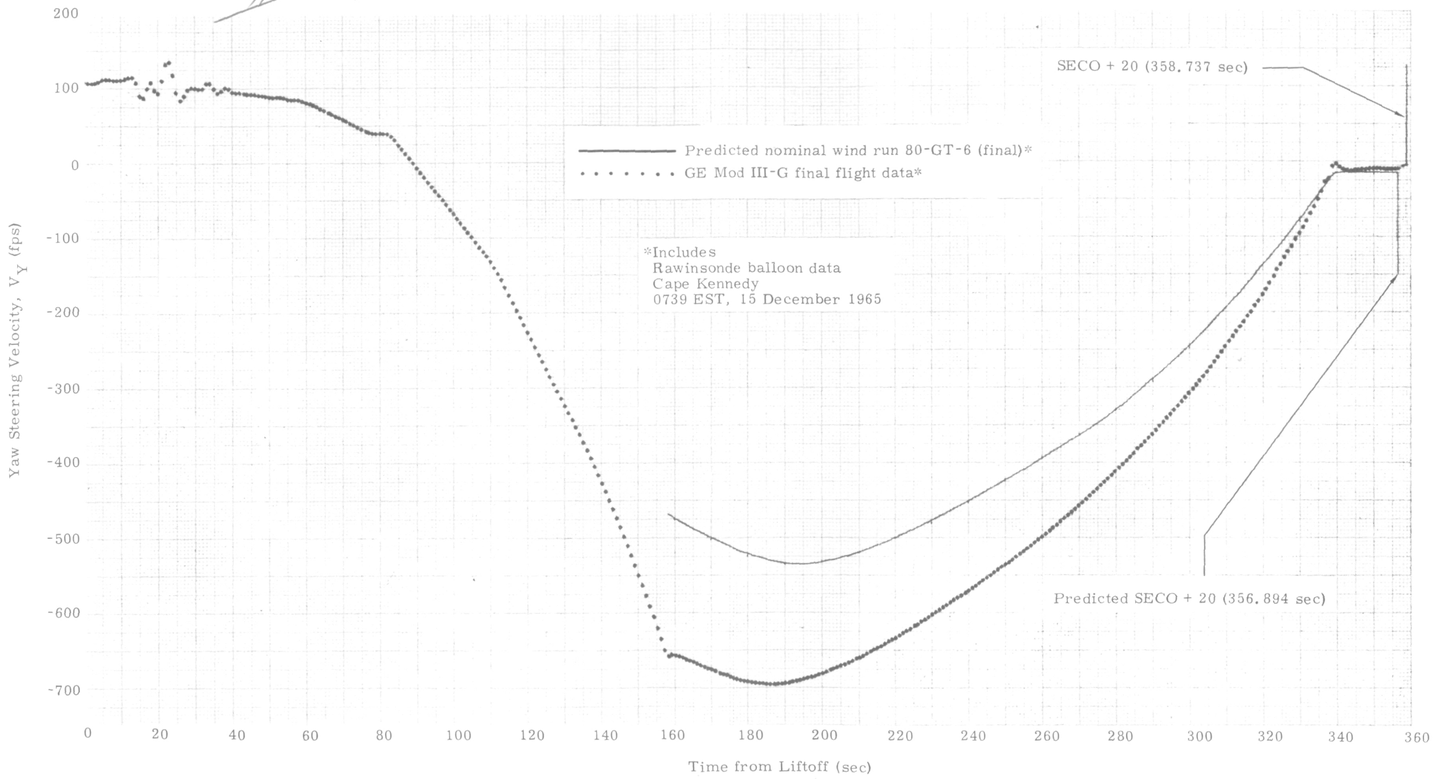

Fig. II-24. Yaw Steering Velocity (V_Y) Versus Time

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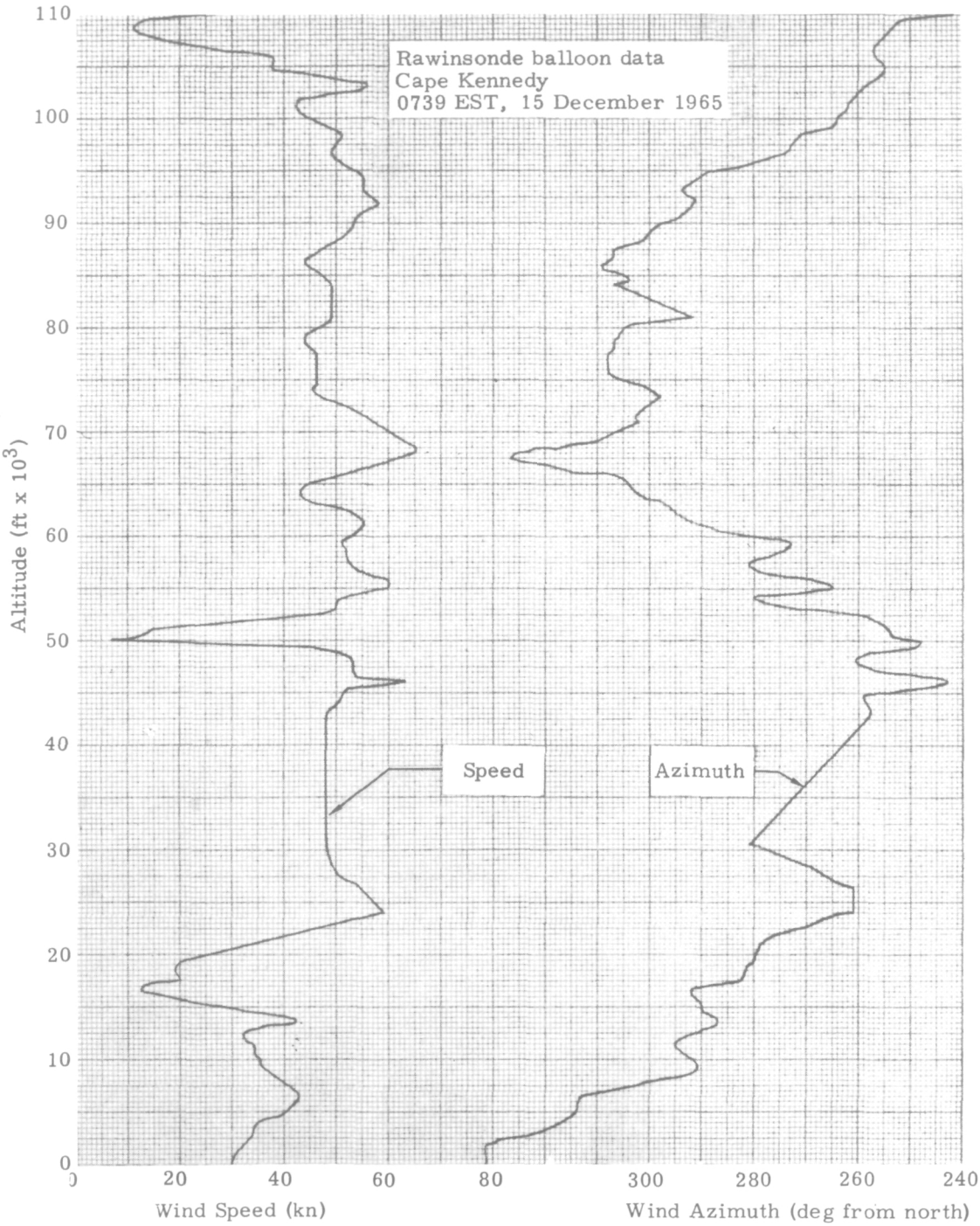

Fig. II-25. Wind Speed and Azimuth Versus Altitude

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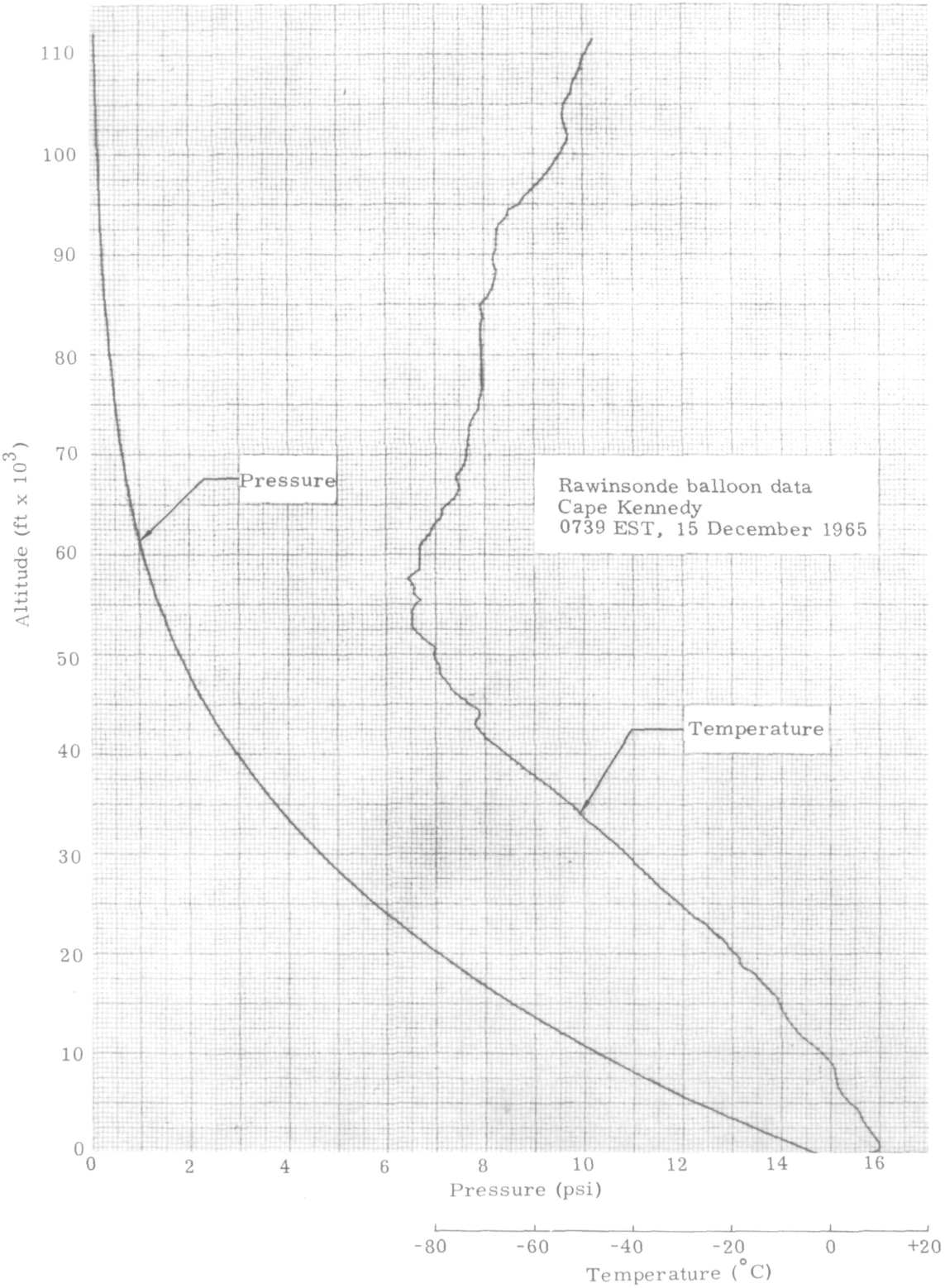

Fig. II-26. Ambient Temperature and Pressure Versus Altitude

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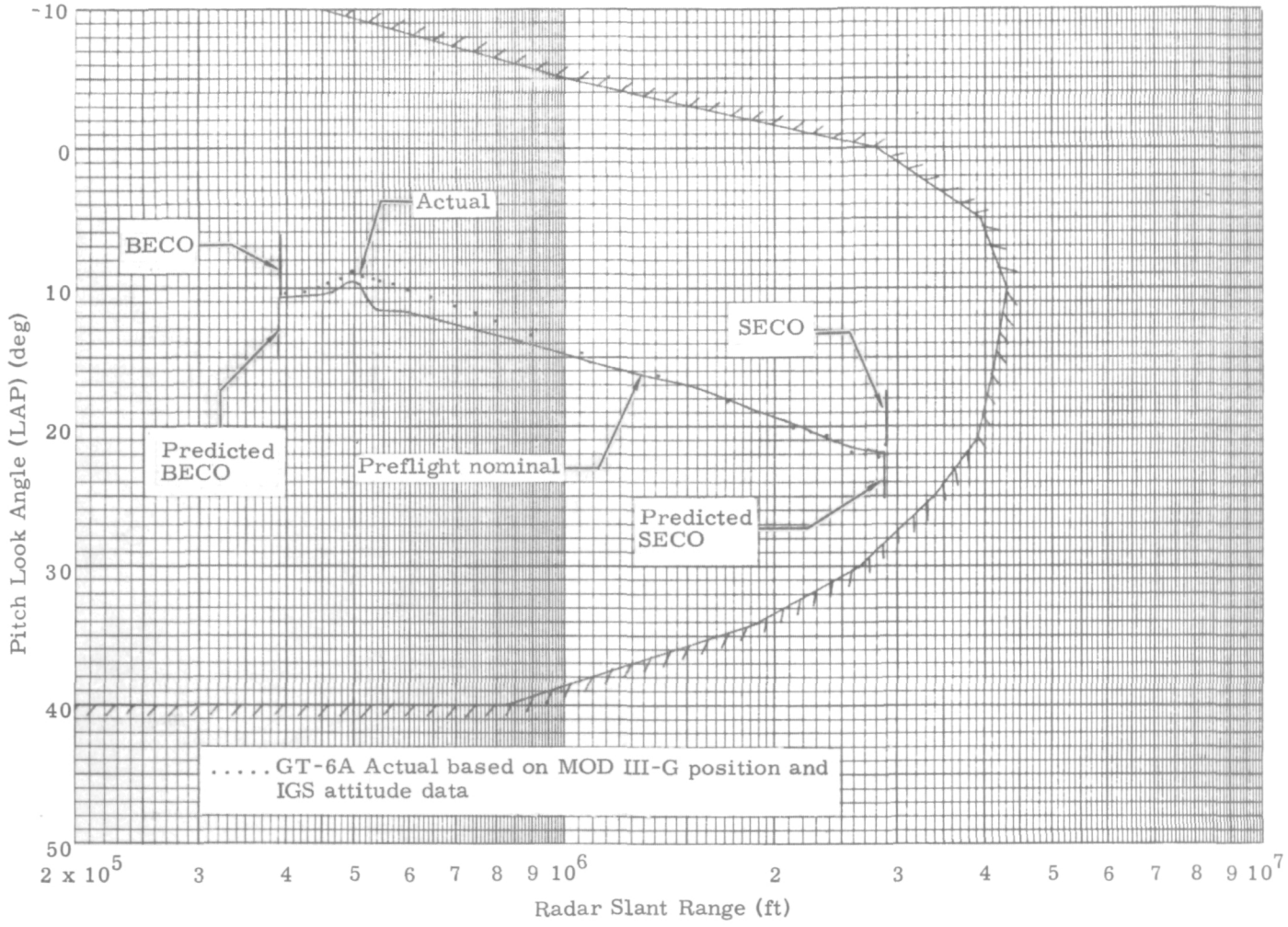

Fig. II-27. Permissible Pitch Look-Angle Envelope

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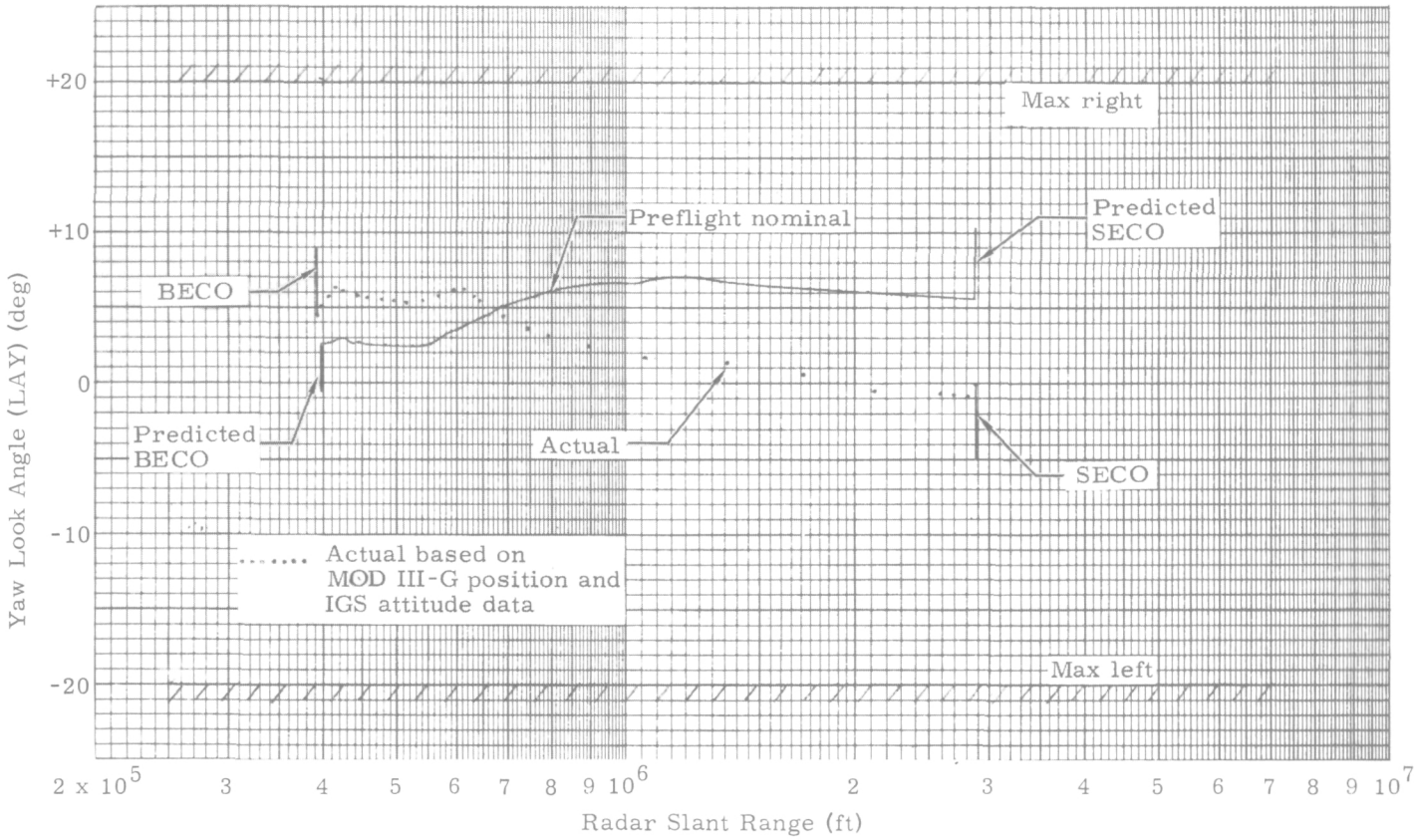

Fig. II-28. Permissible Yaw Look-Angle Envelope

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7. Maximum Dynamic Pressure

The maximum dynamic pressure for the GT-6 trajectory was less than design limits. Table II-9 compares the predicted and observed conditions associated with the maximum dynamic pressure. The predominantly tail-wind environment for this flight in itself reduces the maximum dynamic pressure. A predicted trajectory computation for a no wind condition showed that the maximum dynamic pressure would be 737.5 psi, and the predicted trajectory with T-1 winds, from Table II-9, shows a value of 729.7 psi, verifying the effect of a tail wind. However, the observed maximum dynamic pressure was the same as the predicted no wind value. Therefore, other factors such as engine performance and the TARS pitch programming combined to increase the maximum dynamic pressure slightly compared to the prediction.

TABLE II-9

Trajectory Parameters at Maximum Dynamic Pressure

	Predicted* (nominal)	Observed**
Dynamic pressure (psf)	729.7	737.6
Time from liftoff (sec)	82.8	84.1
Mach number	1.89	1.91
Altitude (ft)	47,650	47,847
Relative flight path angle (deg)	46.28	45.31
Relative wind velocity (fps)	1768	1792
Wind velocity (fps)	92	92
Wind azimuth (deg from north)	281	281
Angle of attack (deg)	0.61	0.86
Angle of sideslip (deg)	0.86	-0.24

*Ref. 10, updated (see footnote to Table II-7)
**Mod III-G 10 pps radar data

8. Angles of Attack and Sideslip

Predicted and observed histories of angles of attack and sideslip during the ascent are shown in Figs. II-12, II-13, II-20 and II-21. The predicted values were obtained from a digital run utilizing wind and atmospheric information obtained from the 0739 EST Rawinsonde sounding. Observed angles of attack and sideslip were derived using the Mod III-G position and velocity information, IGS attitude data and the aforementioned weather data.

B. PAYLOAD CAPABILITY

Propellants remaining onboard after Stage II low level sensor uncover indicated that a burning time margin (BTM) of 2.348 seconds existed to a command shutdown. The total propellant weight margin was 767 pounds, and the corresponding GLV payload capability was 8655 pounds. These values and the predicted nominal and minimum values appear in Fig. II-29. The predicted capability curves were taken from the GLV-6 preflight report (Ref. 13), updated to incorporate the 81.4-degree launch azimuth, yaw steering to correct for the 0.2002-degree wedge angle, revised guidance constants, and the -1.07% pitch and -1.4% roll programmer biases. The predicted propellant weight and burning time margins are based on the difference between these curves and the 7821-pound spacecraft weight.

Real-time payload predictions differed from the predictions shown in Fig. II-29 because extrapolated actual propellant temperatures were used instead of preflight predicted propellant temperatures. The last payload prediction indicated that the minimum payload capability was 275 pounds more than the spacecraft weight, and the nominal payload capability was 878 pounds greater than the spacecraft weight at the predicted launch time. The actual (postflight reconstructed) GLV capability was 834 pounds greater than the spacecraft weight.

C. STAGING

The staging sequence was normal and physical stage separation occurred as planned. The time interval from staging signal ($87FS_2/91FS_1$) to start of Stage II engine chamber pressure (P_{c_3}) rise was 0.667 second. This compares favorably with the nominal expected time of 0.70± 0.08 second. Stage separation occurred 0.015 second following start of P_{c_3} rise.

D. WEIGHT STATEMENT

Table II-10 shows the GT-6 weight history from launch to orbital insertion.

The postflight weight report (Ref. 11) provides the background data for this summary. The report includes a list of dry weight empty changes at ETR and shows a derivation of weight empty from the actual vehicle weighing. Other items covered include the derivation of burnout, BECO, SECO and shutdown weights; weight comparisons with the BLH data; and the center of gravity travel envelope as a function of burn time for the horizontal, vertical and lateral planes.

TABLE II-10

GT-6 Weight Summary

	Weight (lb)			
	Step I	Step II	Step III	Stage Total
Loaded weight	272,925	65,343	7,821 (1)	346,089
Start and grain losses	3,617 (3)			
Trajectory LO weight	269,308	65,343	7,821	342,472
Propellant consumed to BECO	257,533			
Coolant water	0			
Fuel bleed		11		
Weight at BECO	11,775	65,332	7,821	84,928
Shutdown propellant	164			
Stage I burnout	11,611 (2)	65,332 (2)	7,821	84,764
Stage II engine start	11,611	188		
Grain loss		3		
Stage II LO		65,141	7,821	72,962
Propellant consumed to SECO		58,754		
Ablative, covers and coolant water		20		
Stage II at SECO		6,367	7,821	14,188 (4)

TABLE II-10 (continued)

	Weight (lb)			
	Step I	Step II	Step III	Stage Total
Shutdown propellant		136		
Weight at SECO + 20 seconds		6,231	7,821	14,052(4)

(1) Information from NASA-Houston

(2) Includes outage: 834-lb Stage I; 255-lb Stage II

(3) Event: launch bolts blown

(4) Includes 767 lb of usable propellant

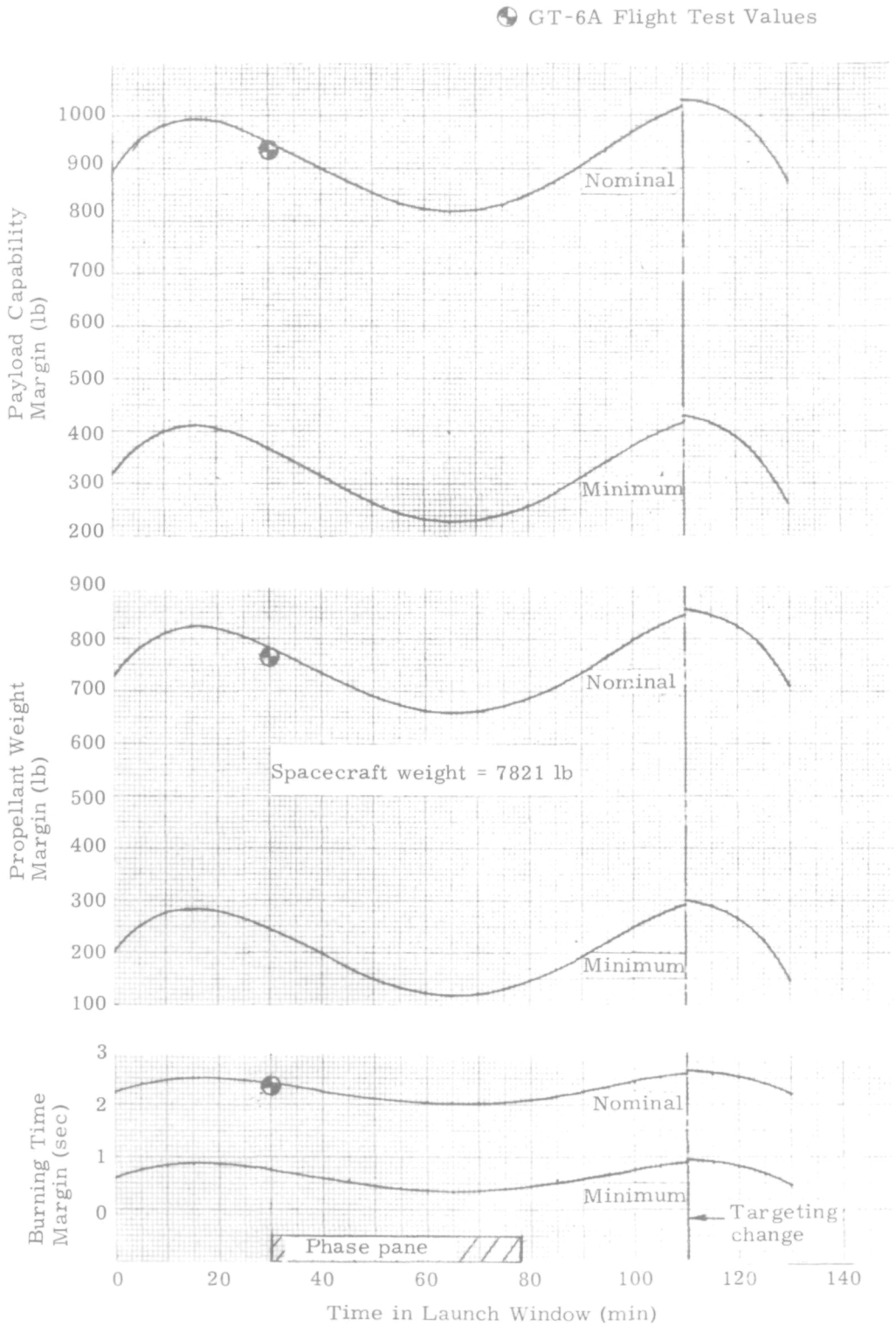

Fig. II-29. Payload Capability

III. PROPULSION SYSTEM

A. LAUNCH ATTEMPT (12 DECEMBER 1965)

The launch of GLV-6 on 12 December 1965 was automatically terminated at $87FS_1$ + 1.158 seconds due to premature separation of the electrical tail plug disconnect 3D1M.

GLV-6 Stage I engine experienced a normal start transient through thrust chamber ignition and into secondary rise until start cartridge burnout and gas generator ignition. S/A 1 start transient appeared nominal and nearly reached equilibrium conditions at the time the shutdown command was initiated. S/A 2 thrust chamber pressure rise was normal through the completion of start cartridge burning, but the subassembly failed to achieve satisfactory bootstrapping and all parameters started to decay abnormally at $87FS_1$ + 1.05 seconds. S/A 2 data showed that sufficient energy was generated by the start cartridge to provide proper bootstrapping operation. Gas generator combustion in S/A 2 apparently was not sustained, and a restriction in propellant flow to the gas generator was the suspected cause.

To determine the exact cause of engine performance decay, the S/A 2 gas generator, fuel and oxidizer bootstrap lines, fuel and oxidizer check valves and strainers were removed for inspection. A small plastic dust cap was found lodged in the gas generator oxidizer injector inlet, which prevented oxidizer flow to the gas generator.

As shown in Figs. III-1 and III-2, both S/A 1 MDTCPS and S/A 2 MDTCPS actuated and de-actuated during the flight attempt. S/A 1 chamber pressure exceeded the TCPS actuation tolerance band (600 to 640 psia); therefore, it is concluded that S/A 1 TCPS had actuated. As shown in Fig. III-1, S/A 2 chamber pressure reached the TCPS actuation limits only momentarily. Both subassembly switches must actuate before TCPS make signal is given, and because this signal was not given for GLV-6 attempt, it was assumed that S/A 2 TCPS did not make.

It was concluded that, had the disconnect malfunction not initiated shutdown, the engine would have received a shutdown command at T + 2.2 seconds due to TCPS not being actuated.

The fuel autogenous system was functioning normally, which is indicated by the fuel pressurant differential pressure switch (FPDPS) actuation at $87FS_1$ + 0.98 second as shown in Fig. III-3. The oxidizer pressurant orifice inlet pressure (POPOI) also shown in Fig. III-3 did not reach the OPPS actuation limits of 360 to 445 psia; hence, OPPS did not actuate. POPOI reached a maximum of only 212 psia.

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III-2-2

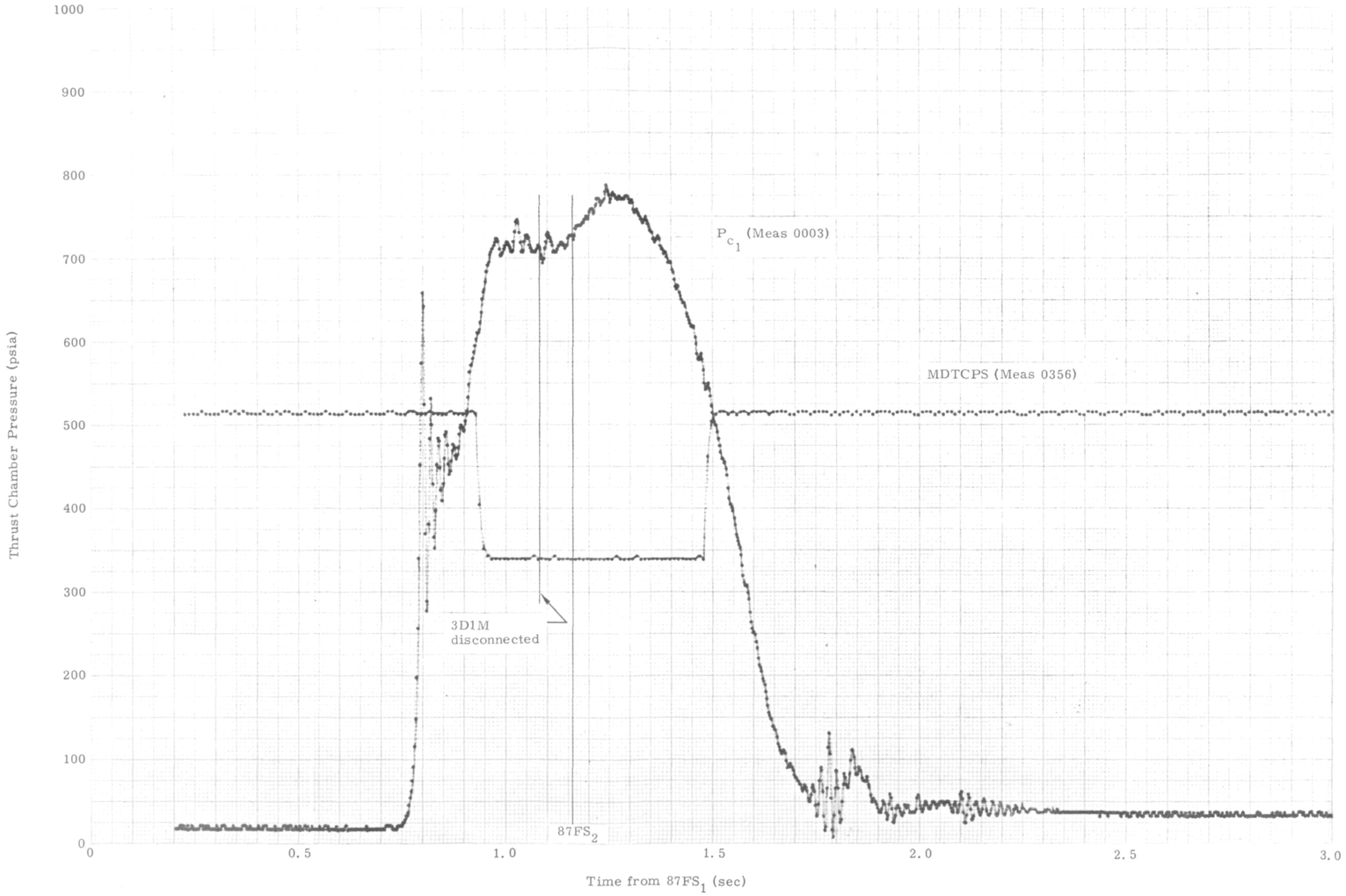

Fig. III-1. Launch Attempt S/A 1 Start Transient

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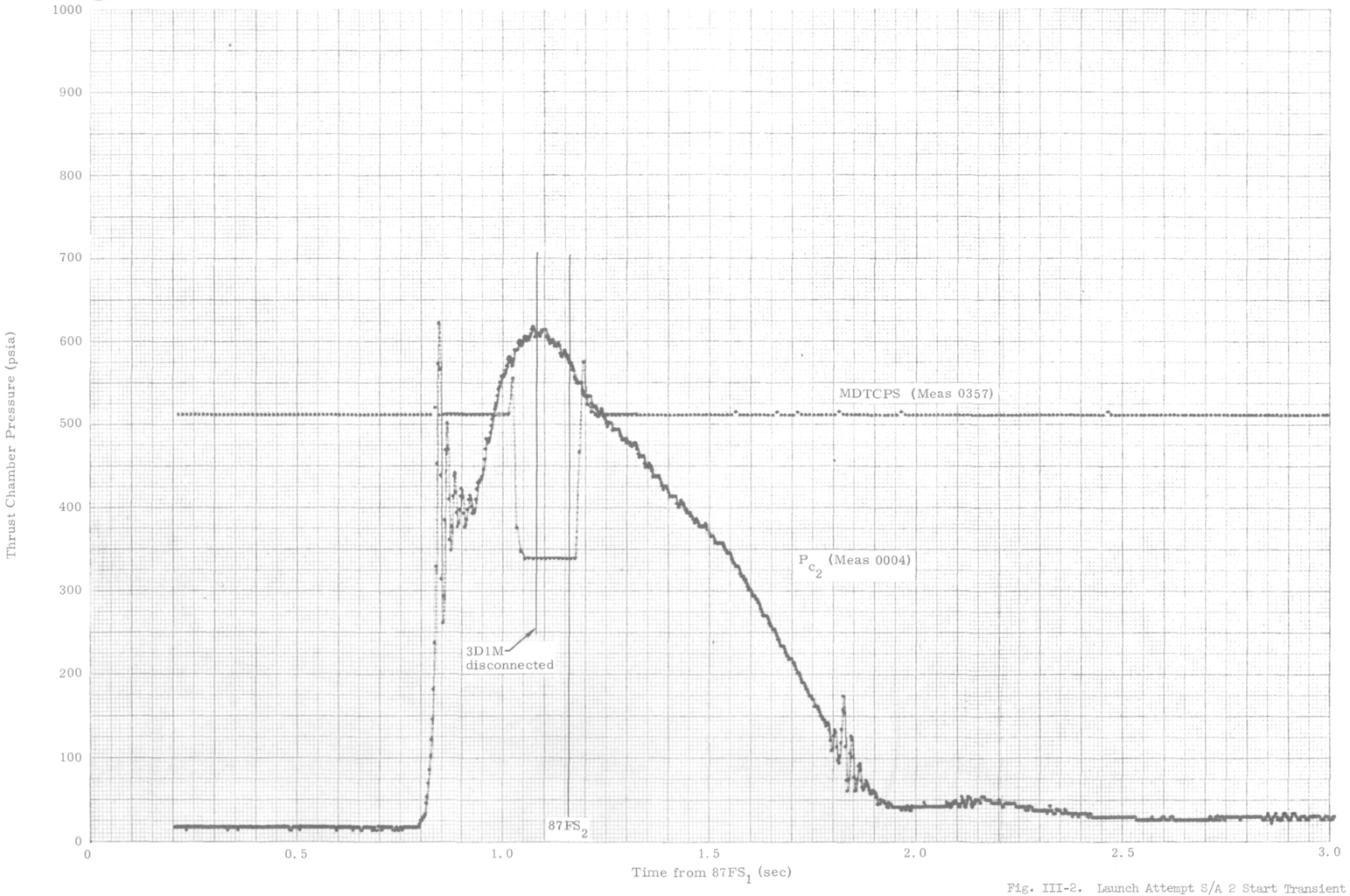

Fig. III-2. Launch Attempt S/A 2 Start Transient

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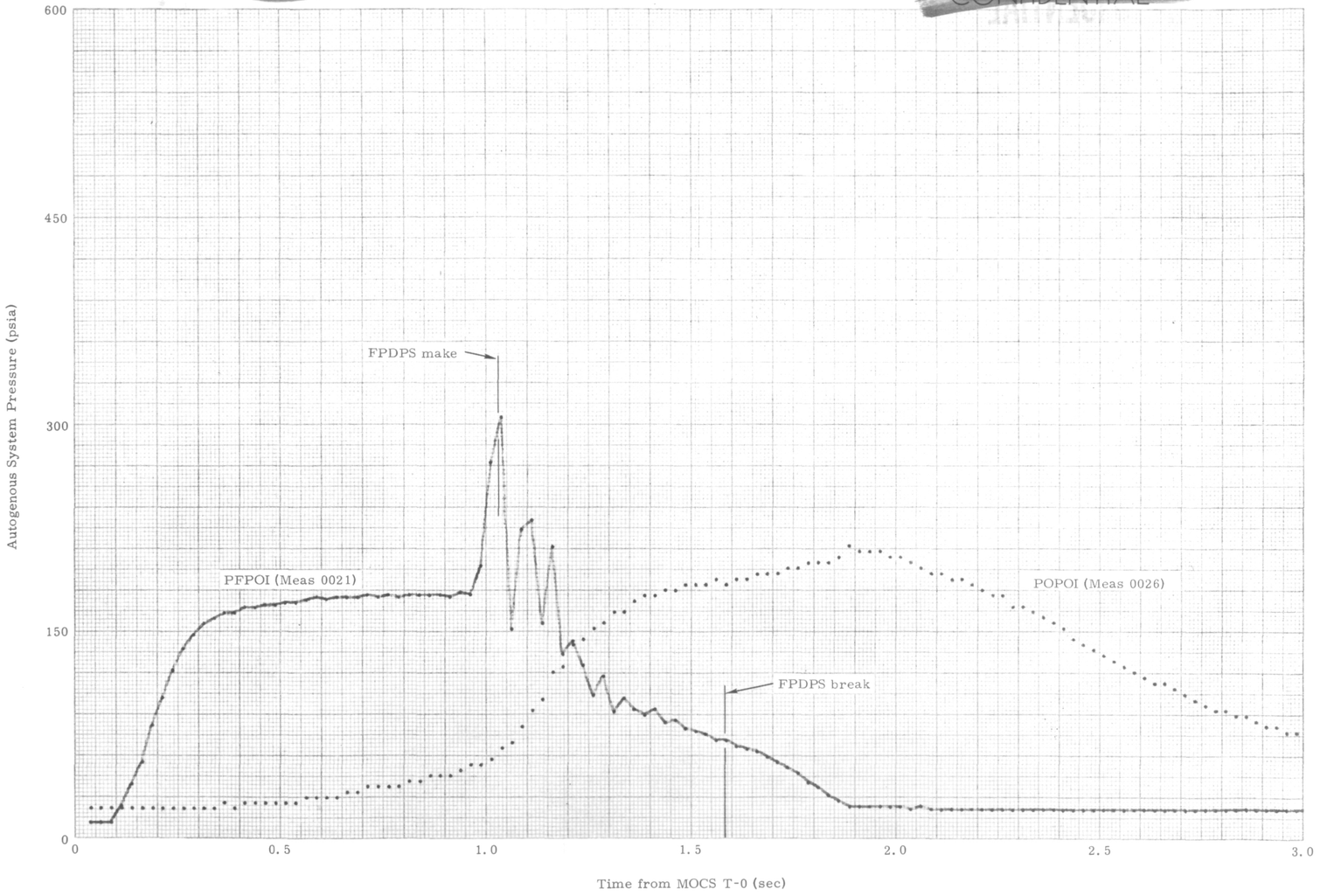

Fig. III-3. Launch Attempt Start Transient Stage I S/A 2

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B. ENGINE SUBSYSTEM

The Stages I and II engines operated satisfactorily throughout the flight, and all launch objectives were met. The Stage I burning time was 160.441 seconds, and shutdown was initiated by oxidizer exhaustion. Stage II engine operation was terminated by guidance command after 181.575 seconds of burning time.

Several anomalies occurred during the flight, none of which adversely affected engine performance. These were:

(1) Subassembly 1 thrust chamber pressure (P_{c_1}, Meas 0003) exhibited unusually high amplitude oscillations throughout Stage I flight. The oscillation frequency was approximately 48 cps with a superimposed beat frequency of 3 cps. The maximum P_c amplitude was 60 psi peak-to-peak. A review of acceptance test data for this engine showed no similar occurrence.

(2) Subassembly 3 (Stage II engine) oxidizer discharge pressure (P_{OD-3}, Meas 0509) also exhibited oscillations and beats during Stage I flight. Maximum amplitude was 90 psi peak-to-peak. The parameter appeared normal during Stage II flight. This anomaly is attributed to RF interference to the transducer.

(3) Four post-SECO disturbances were noted. The first two appeared only on the ± 1/2 g accelerometer and were of very low magnitudes (< 0.02 g peak-to-peak). The latter two were also reflected in the rate gyros and actuator motion traces: No. 3 occurred at SECO + 17.5 seconds at 0.089 g peak-to-peak, and No. 4 occurred at SECO + 28 seconds at 0.093 g peak-to-peak. None of the disturbances were reflected in the engine parameters.

1. Stage I Engine (YLR87-AJ-7, S/N 1006)

a. Configuration and special procedures

The only significant difference in hardware on GLV-6 from that of GLV-7 was the reduction in size of the oxidizer pressurant back pressure orifice from 0.50-inch diameter to 0.46-inch diameter. This change, Aerojet ECP-183, was made in order to increase the confidence level that the oxidizer pressurant pressure switch (automatic engine shutdown switch) would actuate prior to T_0 + 2.2 seconds. The

III-6-2 CONFIDENTIAL

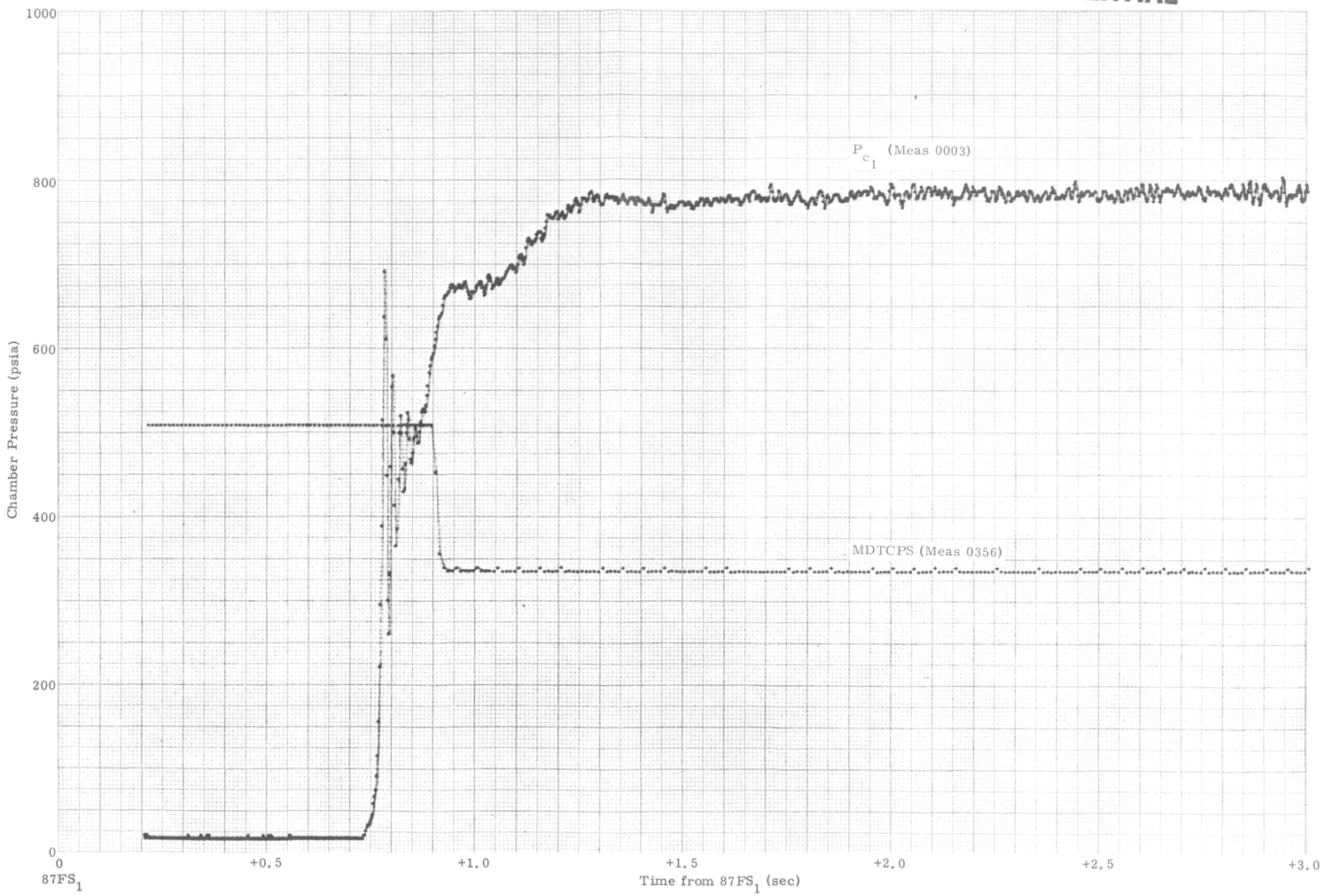

Fig. III-4. S/A 1 Start Transient

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P_{c_1} transducer was wrapped with additional thermal insulation to evaluate the effects of thermal environment on transducer drift.

Following the 25 October 1965 countdown and during the time of GLV-6 bonded storage, all prevalves were removed. New prevalves were installed prior to the 12 December 1965 launch attempt.

b. Start transient

The S/A 1 and S/A 2 thrust chamber start transients were normal as shown in Figs. III-4 and III-5. The ignition spikes indicated 89% of rated thrust for both S/A 1 and S/A 2, which is above the engine model specification allowable (75%). However, the Gemini P_c instrumentation has characteristically shown undamped oscillations which obscure the true transient performance and prevent accurate determination of the ignition spikes. Significant start events are presented in Table III-1.

TABLE III-1

Stage I Engine Start Parameters

Parameter	S/A 1	S/A 2
FS_1 to initial P_c rise (sec)	0.736	0.751
P_c ignition spike (psia)	692	696
P_c step (psia)	465	445
P_c overshoot (psia)	None	None

c. Steady-state performance

Stage I engine flight performance agreed closely with the preflight prediction. Flight integrated average performance parameters were within 1.0% of the preflight predicted.

Engine performance was calculated from measured flight data with the Martin-Baltimore PRESTO program and used the Stage I thrust coefficient relationship as modified by Martin. The modification increased thrust and specific impulse approximately 3400 pounds and 2.0 seconds, respectively, above the values calculated with the Aerojet thrust coefficient relationship. The Martin-modified thrust coefficient also was used in the preflight predictions.

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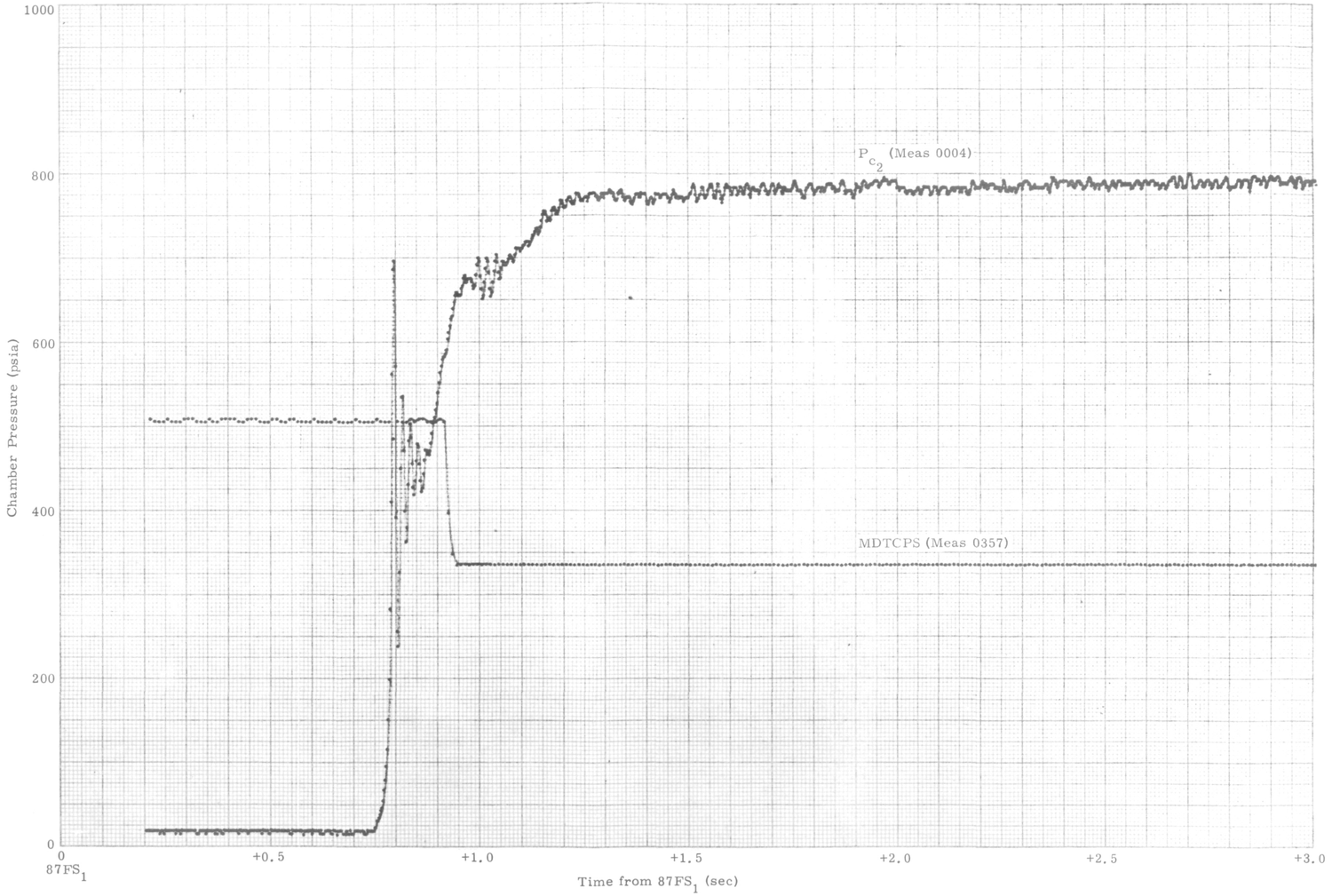

Fig. III-5. S/A 2 Start Transient

The Stage I engine average flight performance, integrated from liftoff to $87FS_2$, is compared with the preflight prediction in Table III-2.

TABLE III-2

Predicted and Flight Performance Comparison--Stage I Engine

Parameter	Preflight* Average	Flight* Average	Difference (%)
Thrust, engine (lb)	455,742	453,793	-0.43
Specific impulse, engine (sec)	276.85	277.27	+0.15
Mixture ratio, engine	1.9248	1 9381	+0.69
Oxidizer flow rate, overboard (lb/sec)	1082.97	1079 29	-0.34
Fuel flow rate, overboard (lb/sec)	563.17	557 38	-1 03

*Martin-Baltimore modified thrust coefficient relationship used

Engine performance calculated throughout the Stage I flight is presented in Fig. III-6. The preflight prediction is also shown for comparison.

The S/A 1 thrust chamber pressure transducer was wrapped with four times the normal amount of thermal insulation. The extra insulation was used on the transducer to verify that the P_c transducer drift was due to thermal effects and to confine the drift to acceptable limits Reconstructed data showed that the P_{c_2} transducer, which had normal insulation, began to drift at approximately $87FS_1$ + 80 seconds and drifted approximately -2% (normal for S/A 2 as established from previous Gemini flights) P_{c_1} data showed no negative drift, verifying the theory that P_c drift was due to the excessive thermal environment.

Stage I engine flight performance calculated at the $87FS_1$ + 55 second time slice and corrected to standard inlet conditions is shown in Table III-3. This is compared to the acceptance test and the predicted flight performance at standard inlet conditions and the nominal time as used

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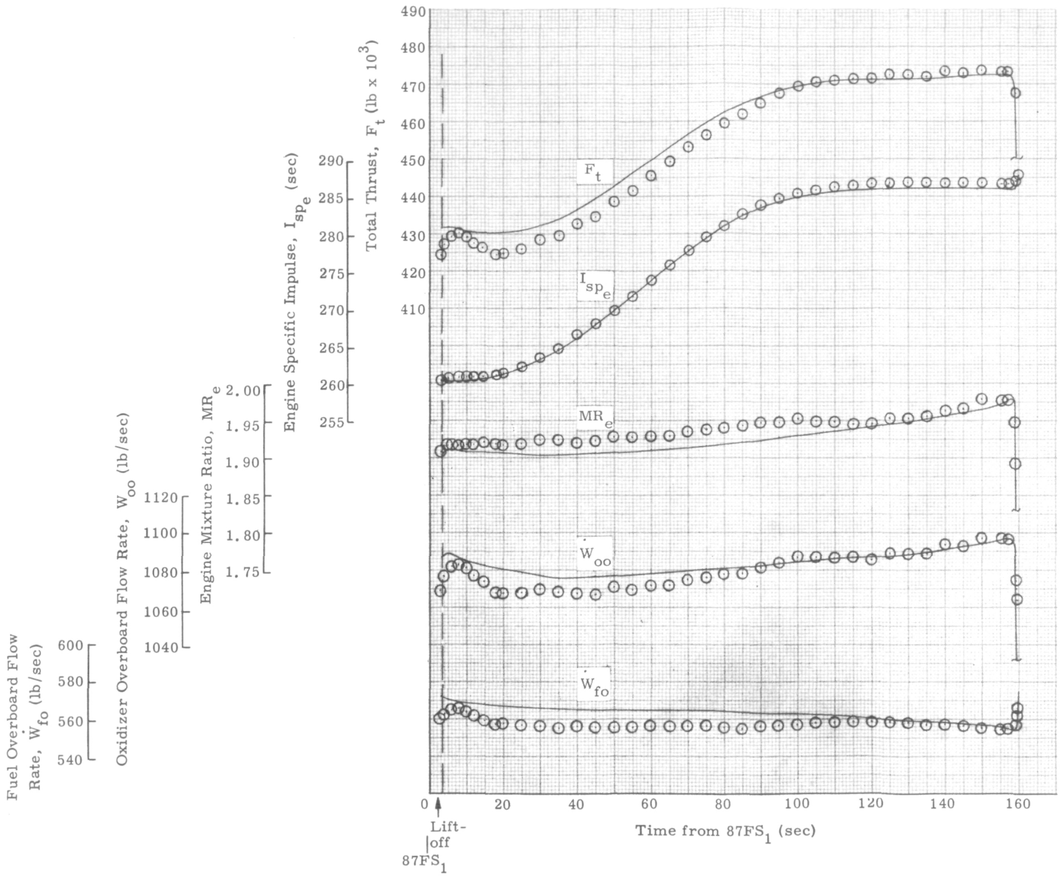

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Average Engine Performance Integrated from Liftoff to $87FS_2$

Symbol	Preflight Prediction	Flight Average
F_t(lb)	455.742	453.793
I_{sp_e} (sec)	276.85	277.27
MR_e	1.9248	1.9381
$\dot{W}_{oo}$(lb/sec)	1082.97	1079.29
$\dot{W}_{fo}$(lb/sec)	563.17	557.38

——— Preflight prediction
○ Flight performance

Fig. III-6. Stage I Engine Flight Performance

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in the preflight prediction. The predicted flight performance at standard conditions was obtained by modifying the nominal acceptance test data for a 4850-pound acceptance-to-flight thrust growth obtained from analysis of previous Titan II and GLV flights.

TABLE III-3

Stage I Performance Corrected to Standard Inlet Conditions at $87FS_1$ + 55 Seconds

Parameter	Acceptance Test*	Predicted Flight (including 4850-lb thrust growth)*	Flight Performance*
Thrust, engine (lb)	427,652	432,502	427,396
Specific impulse, engine (sec)	260.15	260.15	260.02
Mixture ratio, engine	1.9339	1.9339	1.9650
Oxidizer flow rate, overboard (lb/sec)	1083.20	1095.49	1088.98
Fuel flow rate, overboard (lb/sec)	560.65	567.00	554.70

*Martin-Baltimore modified thrust coefficient relationship used.

d. Shutdown transient

Stage I engine shutdown was initiated by oxidizer exhaustion. Figures III-7 and III-8 show S/A 1 and S/A 2 chamber pressure decays. All other engine parameters were normal for an oxidizer exhaustion shutdown. The engine thrust at staging was approximately 24,000 pounds.

Significant events during shutdown are presented in Table III-4.

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TABLE III-4

Stage I Engine Shutdown Parameters

Parameter	S/A 1	S/A 2
Time from P_c decay to $87FS_2$ (sec)	0.7	0.7
P_c at $87FS_2$ (psia)	200	250
Time from FS_2 to data dropout (sec)	0.71	0.71
P_c at data dropout (psia)	50	48

e. Engine malfunction detection system (MDS)

The Stage I engine MDS operated satisfactorily and within specified limits throughout the flight. Figures III-4 and III-5 illustrate response times and actuation levels of the malfunction detection thrust chamber pressure switches (MDTCPS) during engine start for S/A 1 and S/A 2, respectively. Figures III-7 and III-8 show deactuation times and levels during shutdown for S/A 1 and S/A 2, respectively.

A summary of the operating characteristics of the switches is tabulated in Table III-5.

TABLE III-5

Stage I MDTCPS Operation

	Actuation		Deactuation	
Switch	Time (sec)	Pressure (psia)	Time (sec)	Pressure (psia)
S/A 1	FS_1 + 0.900	585	FS_2 - 0.047	550
S/A 2	FS_1 + 0.920	575	FS_2 - 0.045	530

Specification Requirements	
Actuation	540 to 600 psia
Deactuation	585 to 515 psia

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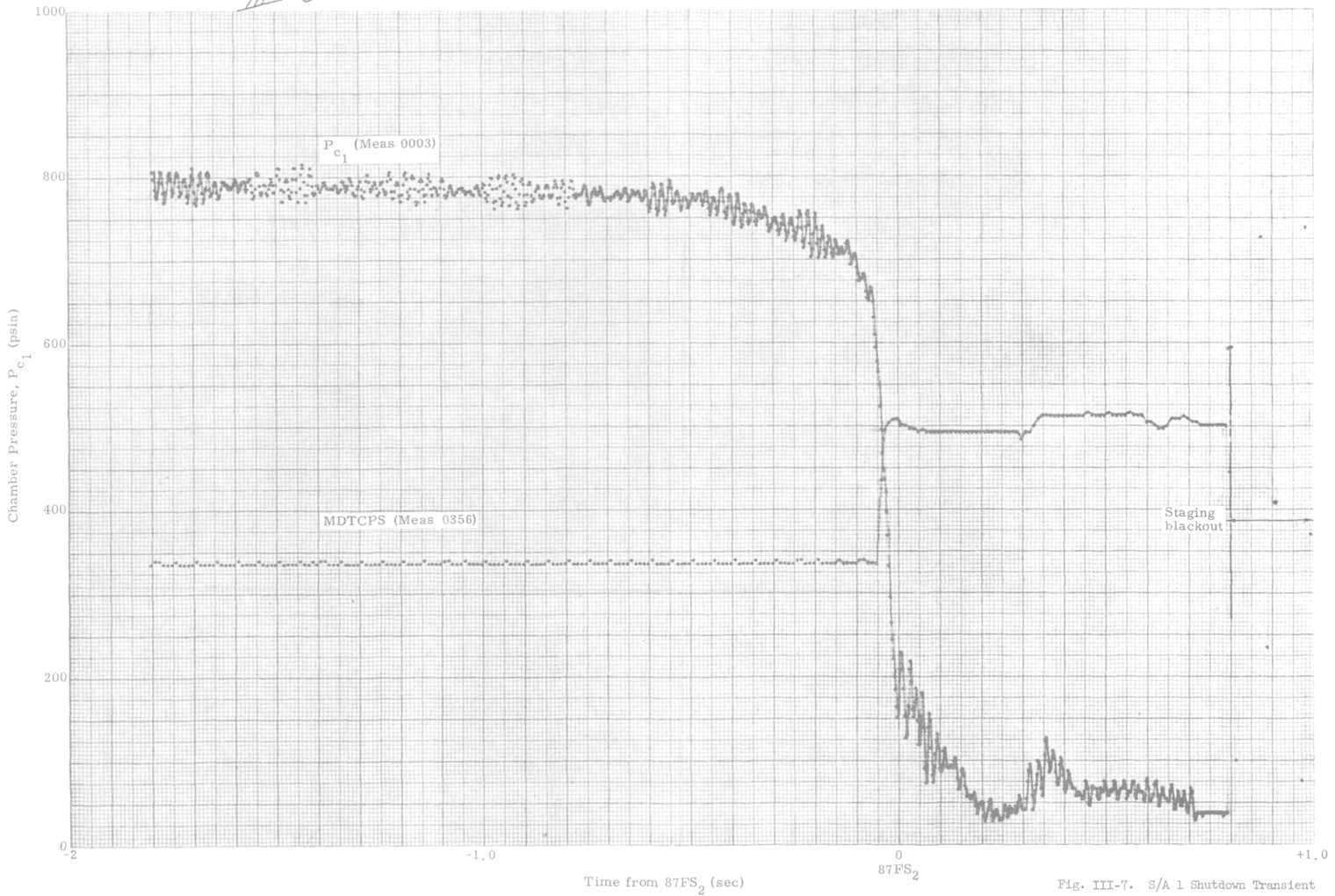

Fig. III-7. S/A 1 Shutdown Transient

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Fig. III-8. S/A 2 Shutdown Transient

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f. Engine prelaunch malfunction detection system (PMDS)

All PMDS switches actuated within specified actuation times and pressures as shown in Table III-6. As a result of the later than expected OPPS actuation on GLV-7 launch, the oxidizer pressurant back pressure orifice diameter was changed from 0.50 inch to 0.46 inch. The smaller back pressure orifice increased the oxidizer pressurant orifice inlet pressure (POPOI); consequently, the OPPS actuated earlier than on previous Gemini flights and substantially earlier than the interrogation time (T-0 + 2.2 seconds), as shown in Fig. III-9.

TABLE III-6

Stage I PMDS Operation

	TCPS	OPPS	FPDPS
Actuation time			
Measured time from $87FS_1$ (sec)	0.981	1.578	0.912
Measured time from T_0 (sec)	1.043	1.640	0.974
Required time (sec)*	T+2.2	T+2.2	T+2.2
Actuation pressure			
Measured (psia)	**	424	**
Required (psia)	600 to 640	360 to 445	46 to 79 (psid)

*The shutdown timers start from T_0.

**Not instrumented.

2. Stage II Engine (YLR91-AJ-7 S/N 2007)

a. Configuration and special procedures

The GLV-6 Stage II engine configuration was identical to that of GLV-7.

During the time of GLV-6 storage, after the 25 October 1965 launch attempt, the Stage II gas generator was removed and returned to Aerojet, Sacramento, for cleaning.

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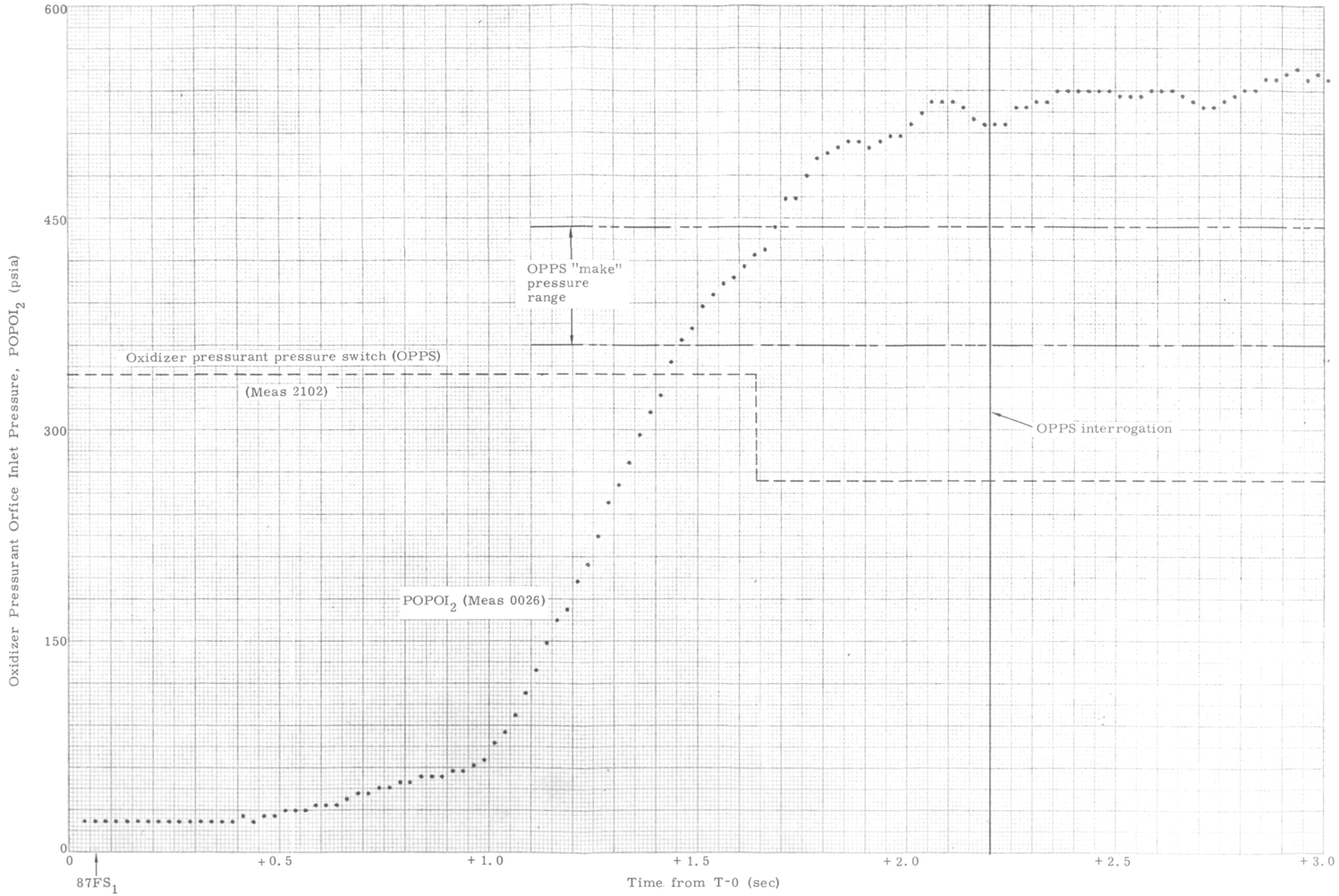

Fig. III-9. S/A 2 Start Transient

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After the 12 December 1965 launch attempt, Stage II recycle requirements were based on Aerojet Engine Test Directive 2.1-3.4E. The TPA oil was changed as a precautionary measure. The prevalves remained installed, and reloading the propellants directly on the thrust chamber valves had no adverse effect on the engine system.

b. Start transient

Stage II engine start transient was normal, as illustrated by the thrust chamber pressure in Fig. III-10. Significant engine start events are presented in Table III-7.

TABLE III-7

Stage II Engine Start Parameters

Parameter	Flight Performance
FS_1 to initial P_{c_3} rise (sec)	0.651
P_{c_3} ignition spike (psia)	699
P_{c_3} step (psia)	495
P_{c_3} overshoot (psia)	Not available*

*Staging blackout period.

c. Steady-state performance

Stage II engine steady-state flight performance was satisfactory throughout flight and agreed closely with preflight predictions. The

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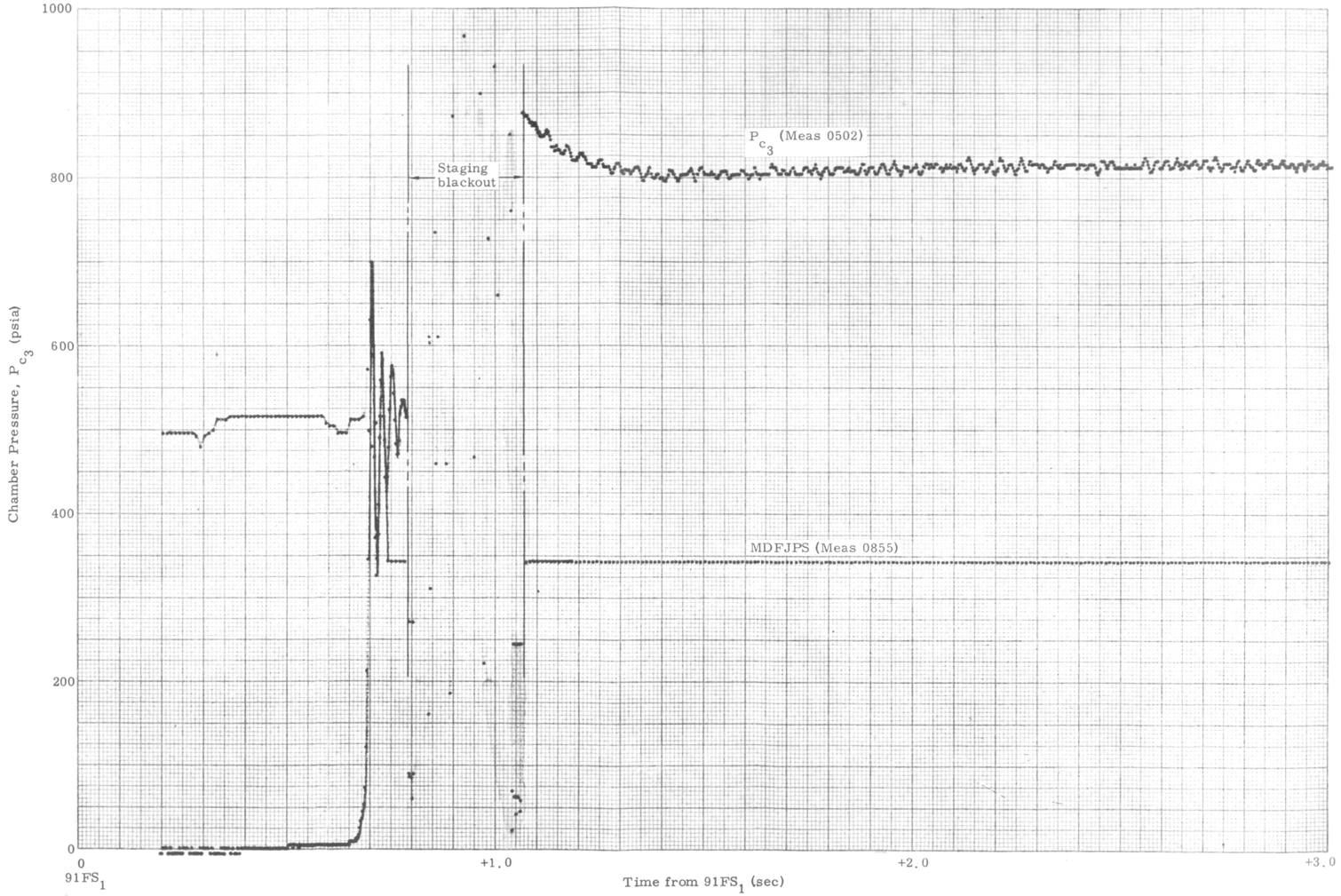

Fig. III-10. S/A 3 Start Transient

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average Stage II engine performance integrated over steady-state operation (from FS_1 + 1.2 seconds to $91FS_2$) is compared to the preflight prediction in Table III-8.

The engine flight performance calculated with the Martin-Baltimore PRESTO program is shown in Fig. III-11 as a function of time from $91FS_1$. The preflight prediction is also presented for comparison.

Engine flight performance corrected at the $91FS_1$ + 55 second time slice to standard inlet conditions is shown in Table III-9. This is compared with the acceptance test and the predicted flight performance at standard inlet conditions and the nominal time as used in the preflight prediction. The predicted flight performance at standard inlet conditions was obtained by adjusting the nominal acceptance test data for a 900-pound acceptance-to-flight thrust growth obtained from analysis of previous Titan II and GLV flights.

d. Shutdown transient

Stage II engine shutdown was initiated by guidance command with $91FS_2$ occurring at $91FS_1$ + 181.575 seconds. The calculated shutdown impulse from $91FS_2$ + 20 seconds was 36,170 lb-sec; predicted impulse was 37,000 ± 7000 lb-sec. The impulse obtained from the ± 10 g accelerometer and illustrated by the P_{c_3} decay in Fig. III-12 was 24,255 lb-sec, using an average spacecraft/Stage II weight of 14,203 pounds. This was for the time interval from $91FS_2$ to $91FS_2$ + 0.687 second. Impulse from $91FS_2$ + 0.687 second to $91FS_2$ + 20 seconds was 11,915 lb-sec, utilizing the ± 0.5 g accelerometer data and an average weight of 14,135 pounds. This thrust tailoff is illustrated in III-13.

Thrust at SECO + 20 seconds was estimated at 30 pounds (below the specified maximum of 60 pounds).

e. Engine malfunction detection systems

The Stage II engine MDS operated satisfactorily throughout flight. Figures III-10 and III-12 illustrate the response times and chamber pressure correlation of the malfunction detection fuel injector pressure

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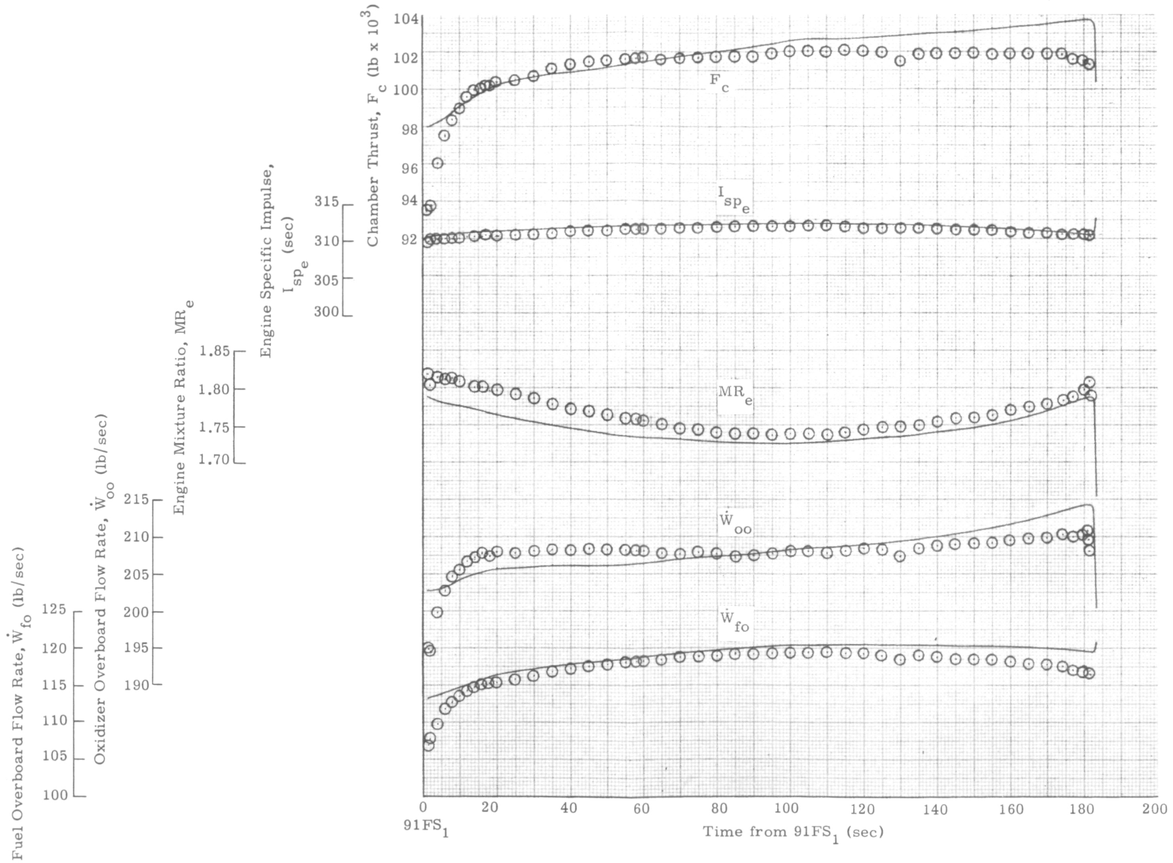

Average Engine Performance Integrated from First Steady-State to 91FS$_2$

Symbol	Preflight Prediction	Flight Average
F_c (lb)	101954	101334
I_{sp_e} (sec)	311.54	311.09
MR_e	1.7448	1.7628
$\dot{W}_{oo}$ (lb/sec)	208.19	208.00
$\dot{W}_{fo}$ (lb/sec)	119.06	117.74

——— Preflight prediction

⊙ Flight performance

Fig. III-11. Stage II Engine Flight Performance

TABLE III-8

Predicted and Average Stage II Engine Performance

Parameter	Preflight Predicted Average	Flight Average	Difference (%)
Thrust, chamber (lb)	101,954	101,334	-0.61
Specific impulse, engine (sec)	311.54	311.09	-0.14
Mixture ratio, engine	1.7448	1.7628	+1.03
Oxidizer flow rate, overboard (lb/sec)	208.19	208.00	-0.09
Fuel flow rate, overboard (lb/sec)	119.06	117.74	-1.11

TABLE III-9

Stage II Engine Performance Corrected to Standard Inlet Conditions at $91FS_1$ + 55 Seconds

Parameter	Acceptance Test	Predicted Flight (including 900-lb thrust growth)	Flight Performance
Thrust, chamber (lb)	100,033	100,933	101,198
Specific impulse, engine (sec)	310.91	310.91	310.76
Mixture ratio, engine	1.7736	1.7736	1.7810
Oxidizer flow rate, overboard (lb/sec)	205.91	207.76	208.72
Fuel flow rate overboard (lb/sec)	115.84	116.88	116.93

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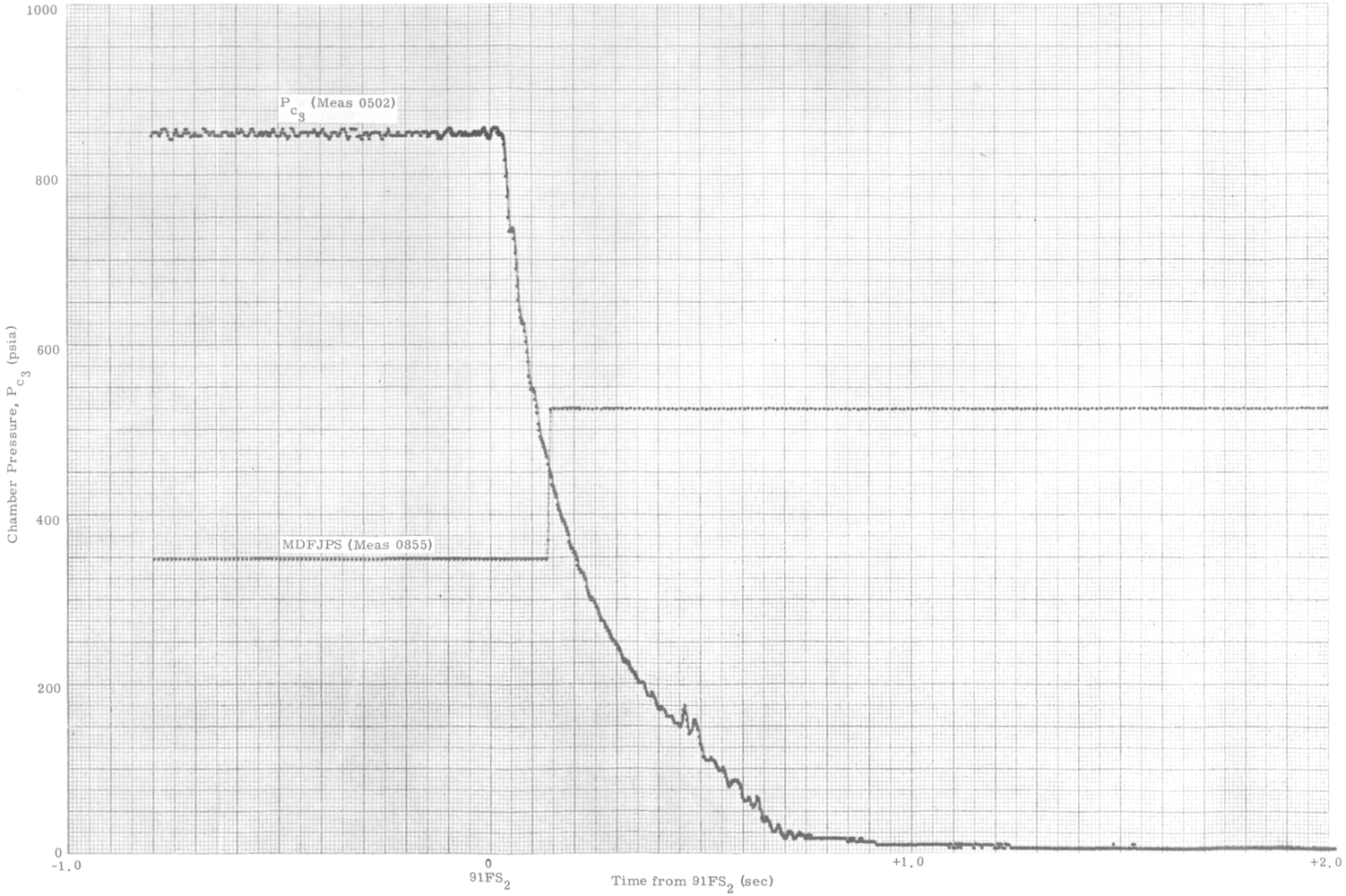

Fig. III-12. S/A 3 Shutdown Transient

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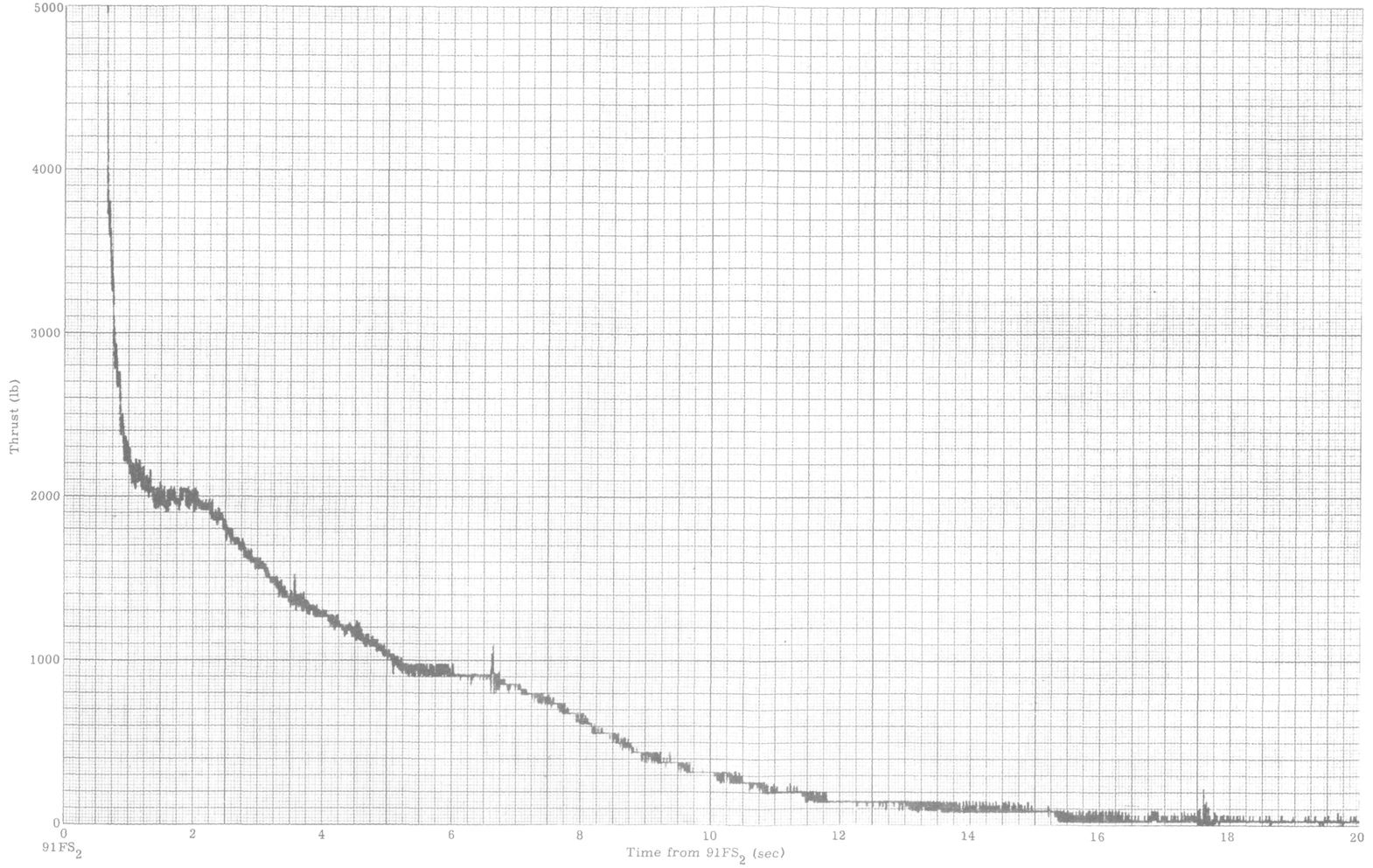

Fig. III-13. Stage II Engine Thrust Tail-Off

switches (MDFJPS) during the start and shutdown transients, respectively. The fuel injector pressure is not instrumented and, therefore, is not available. A summary of the significant switch parameters is presented in Table III-10.

TABLE III-10

Stage II MDS Operation

Parameter	
Actuation time (sec)	$91FS_1 + 0.737$
P_c at actuation (psia)	520
Deactuation time (sec)	$91FS_2 + 0.140$
P_c at deactuation (psia)	450

C. PROPELLANT SUBSYSTEM

1. Propellant Loading

a. Loading procedure

Five propellant loadings were performed on GLV-6, consisting of the RTP and WMSL exercises, two launch attempts and the actual launch. (See Table III-11.)

TABLE III-11

GLV-6 Loadings

Operation	Description	Date
RTP	Dual loading	28 September 1965
WMSL	Dual loading	7 October 1965
1st launch attempt	Dual loading	24 October 1965
2nd launch attempt	Dual loading	12 December 1965
Launch	Dual loading	15 December 1965

All loadings were made using the tandem flowmeter system installed after the launch of GT-5. No serious ground or airborne hardware

problems occurred during the five propellant loadings; however, a number of flowmeters were removed following the WMSL and again following the 12 October 1965 launch attempt.

The check calibration results of the flowmeters removed from the system are presented in Table III-12. These data have been applied to the differences observed between flowmeter and tab run wherever applicable.

TABLE III-12

Flowmeter Verification Results

Test After Which Meter Was Checked	Meter Position	Meter Serial No.	Calibration Facility	Verification Facility	Meter Error
WMSL	Stage I fuel	202146	Martin-Denver	Martin-Denver	+0.166
WMSL	Stage I fuel	199172	Martin-Denver	Martin-Denver	+0.58
WMSL	Stage II fuel	206361	Martin-Denver	Martin-Denver	+0.39
WMSL	Stage II fuel	206361	Martin-Denver	Wyle	+0.14
WMSL	Stage II fuel	199170	Martin-Denver	Martin-Denver	+0.27
1st launch attempt	Stage I fuel	199169	Martin-Denver	Martin-Denver	+0.41
	Stage II fuel	199170	Martin-Denver	Martin-Denver	+0.27

The tab runs used for the five loading operations were derived as follows:

(1) RTP — Obtained from Denver tank calibration data.

(2) WMSL — Obtained from Denver tank calibration data.

(3) First Launch Attempt — Original tab run corrected for RTP and WMSL results including flowmeter verifications after the WMSL.

Stage I Fuel	Stage II Fuel	Stage I Oxidizer	Stage II Oxidizer
No change	-0.31%	+0.14%	No change

(4) Second Launch attempt — Tab runs revised to account for results of first three tests, flowmeter verifications after the first launch attempt and the difference between Denver and Wyle calibration facilities. Changes below are from original tab run.

Stage I Fuel	Stage II Fuel	Stage I Oxidizer	Stage II Oxidizer
+0.04%	-0.30%	-0.11%	-0.23%

(5) Launch — Tab run pounds changed to account for open prevalves; otherwise same as that for second launch attempt.

Tests at the Denver and Wyle calibration facilities have established that, if a fuel or oxidizer flowmeter calibrated at Martin-Denver is assumed to be correct, a corresponding Wyle meter will read about 0.3% higher. It is not known which facility is more nearly correct; however, the launch loading was based on the Martin-Denver calibrated flowmeters being correct. This, in effect, decreased the Wyle calibrated flowmeter/tab run errors by 0.3% and established the least probability of payload loss.

A detailed summary of results of the five propellant loadings made for GLV-6 is shown in Table III 13.

TABLE III-13

Summary of Propellant Load Verification

Loading Event	Tank	Flowmeter				Hi-Lite Temperature (°F)	Difference Between Flowmeter and Actual Tab Run Nom (%) (2)	Differnece Between Flowmeter and First Tab Run (%) (3)	Allowable Tolerance (%)	Average Load-in Flow Rate (gpm)
		Serial No.	PTPS No.	Connected to CP 2500 Counters	Calibration Facility (1)					
RTP	Stage I fuel	202194	441		D	44.0	+0.141	-0.025	±0.3	243
		199172	4411	X	D	44.0	+0.398	-0.192		
	Stage II fuel	206361	445		D	46.9	+0.086	-0.304		121
		199170	4419	X	D	46.9	-0.070	-0.340		
	Stage I oxidizer	204277	442	X	W	48.6	+0.168	+0.168		205
		206359	4412		W	48.6	+0.318	+0.318		
	Stage II oxidizer	206168	446		W	49.8	+0.111	+0.111		97
		199173	4420	X	W	49.8	+0.152	+0.152		
WMSL	Stage I fuel	202146	441	X	D	29.9	+0.358	+0.192		230
		199172	4411		D	29.9	+0.645	+0.065		
	Stage II fuel	206361	445		D	30.8	+0.174	-0.216		109
		199170	4419	X	D	30.8	+0.008	-0.262		
	Stage I oxidizer	204277	442	X	W	26.9	+0.082	+0.082		175
		206359	4412		W	26.9	+0.084	+0.084		
	Stage II oxidizer	199168	446		W	29.4	+0.032	+0.032		85
		199173	4420	X	W	29.4	-0.109	-0.109		
First launch attempt	Stage I fuel	199169	441		D	34.5	+0.584	+0.174		
		202146	4411	X	D	34.5	+0.006	+0.006		
	Stage II fuel	199171	445	X	D	33.8	+0.065	-0.245		
		199170	4419		D	33.8	+0.170	-0.410		
	Stage I oxidizer	SPOOL	442		-	36.2	--	--		
		202164	4412	X	W	36.2	+0.153	+0.293		
	Stage II oxidizer	199168	446	X	W	37.5	+0.147	+0.147		
		199173	4420		W	37.5	+0.081	+0.081		
Second launch attempt	Stage I fuel	206362	441		W	28.2	+0.469	+0.509		
		202146	4411	X	D	28.2	-0.022	+0.018		
	Stage II fuel	206361	445		W	29.0	+0.477	+0.177		
		204278	4419	X	D	29.0	+0.243	-0.057		
	Stage I oxidizer	206360	442		W	26.6	+0.274	+0.164	±0.1	
		202164	4412	X	W	26.6	+0.378	+0.268		
	Stage II oxidizer	199168	446		W	27.6	+0.355	+0.125		
		199167	4420	X	W	27.6	+0.113	-0.117		
Launch	Stage I fuel	206362	441		W	29.2	+0.497	+0.537	±0.3	
		202146	4411	X	D	29.2	-0.001	+0.039		
	Stage II fuel	206361	445		W	30.0	+0.516	+0.216		
		204278	4419	X	D	30.0	+0.268	-0.032		
	Stage I oxidizer	206360	442		W	29.4	+0.288	+0.178	±0.1	
		202164	4412	X	W	29.4	+0.420	+0.310		
	Stage II oxidizer	199168	446		W	29.8	+0.364	+0.134		
		199167	4420	X	W	29.8	+0.183	-0.047		

(1) W = Wyle laboratories D = Martin-Denver

(2) Actual difference observed during loading, not corrected for meter verification where applicable.

(3) Difference from RTP or WMSL tab run corrected for flow rate and meter verification results where applicable. (Does not include any meter bias.)

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The sequence of propellant loading events is given in Table III-14.

TABLE III-14

GT-6 Launch Propellant Loading Schedule

Event	Time (EST) 14 Dec, 1965			
	Stage I Oxidizer	Stage II Oxidizer	Stage I Fuel	Stage II Fuel
Start prechill	1935	1935	2205	2205
Start load	2012	2013	2230	2230
Hi-lite	2142	2050	2325	2302
Load complete	2155	2108	2333	2307

Mission loads for the oxidizer tanks were obtained by using the K-factor ratio technique. This was in accord with a Martin Company/SSD agreement that an oxidizer flowmeter/tab run error of more than $\pm 0.1\%$ at hi-lite would constitute an out-of-tolerance condition.

A flowmeter-to-tab run comparison is shown in Figs. III-14 and III-15. In each figure, the data are referenced to the tank calibration made at Denver (which is synonymous to the special loading tab run). The data for Wyle calibrated meters are not corrected for the difference between Denver and Wyle facilities. In Fig. III-14 the application of the -0.3% correction to all Wyle results will account for the placement of the launch tab shift.

b. Total propellant loads

Total mission loads for the launch, as determined from flowmeters, are shown in Table III-15. The flowmeter totalizer readings were corrected by subtracting propellant vaporized and propellant remaining in the fill lines. Oxidizer flowmeter loads reflect the use of the K-factor ratio method to obtain mission loads. Total propellant loads as determined by flight verification are also shown in Table III-15. The flight verification loads were calculated from a propellant inventory, using actual level sensor uncover times and tank calibration data to determine flow rates. Total, integrated, in-flight overboard propellant consumption was found using the engine analytical model. Engine start transient consumptions were derived from Aerojet summary reports. Other transient propellant consumptions and pressurization gas weights were calculated from flight data (Tables III-36 and III-37).

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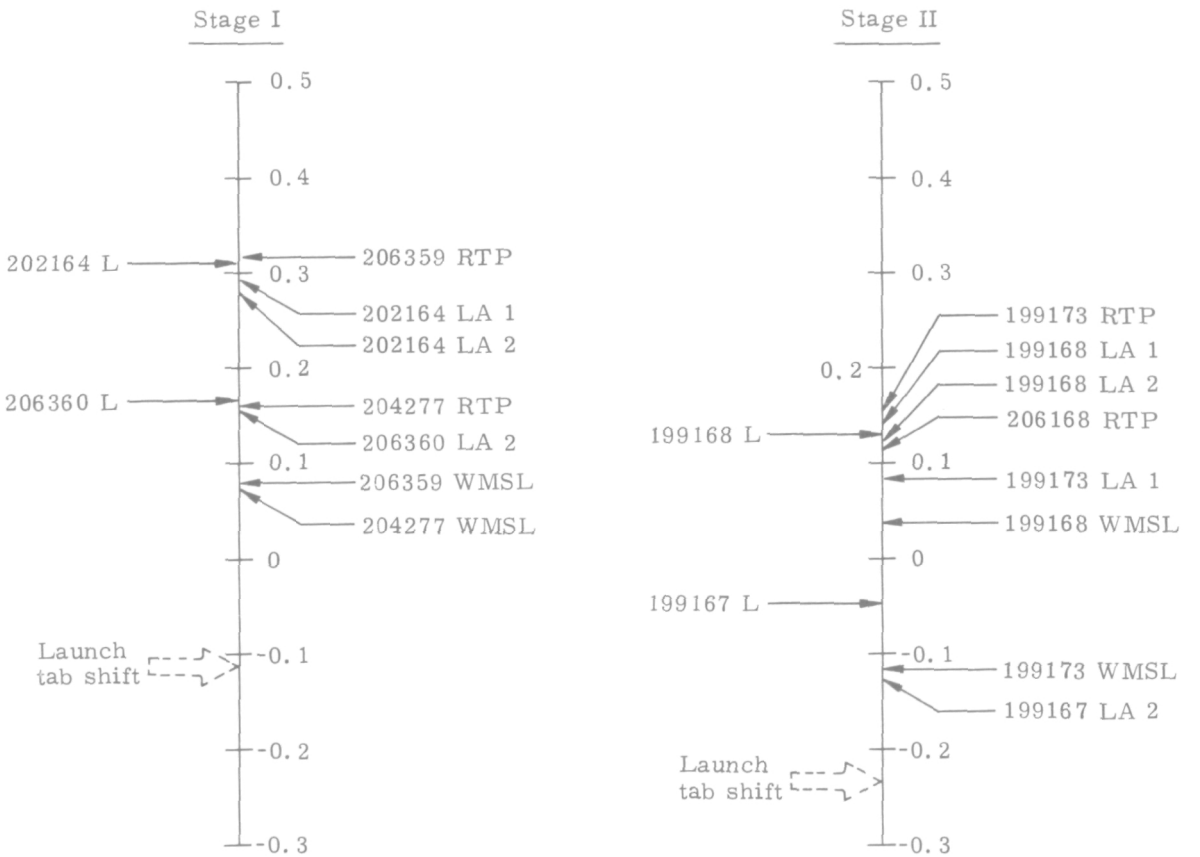
Data are corrected for flow rate and meter verification results, where available. All data are referenced to original tank calibration and represent the percent error of the flowmeter result at hi-lite from the original calibration.



Note: All meters are Wyle calibrated

LEGEND

RTP = first loading
WMSL = wet mock
LA 1 = 25 October 1965 launch attempt
LA 2 = 12 December 1965 launch attempt
L = launch

Fig. III-14. GLV-6 Loading Summary--Oxidizer

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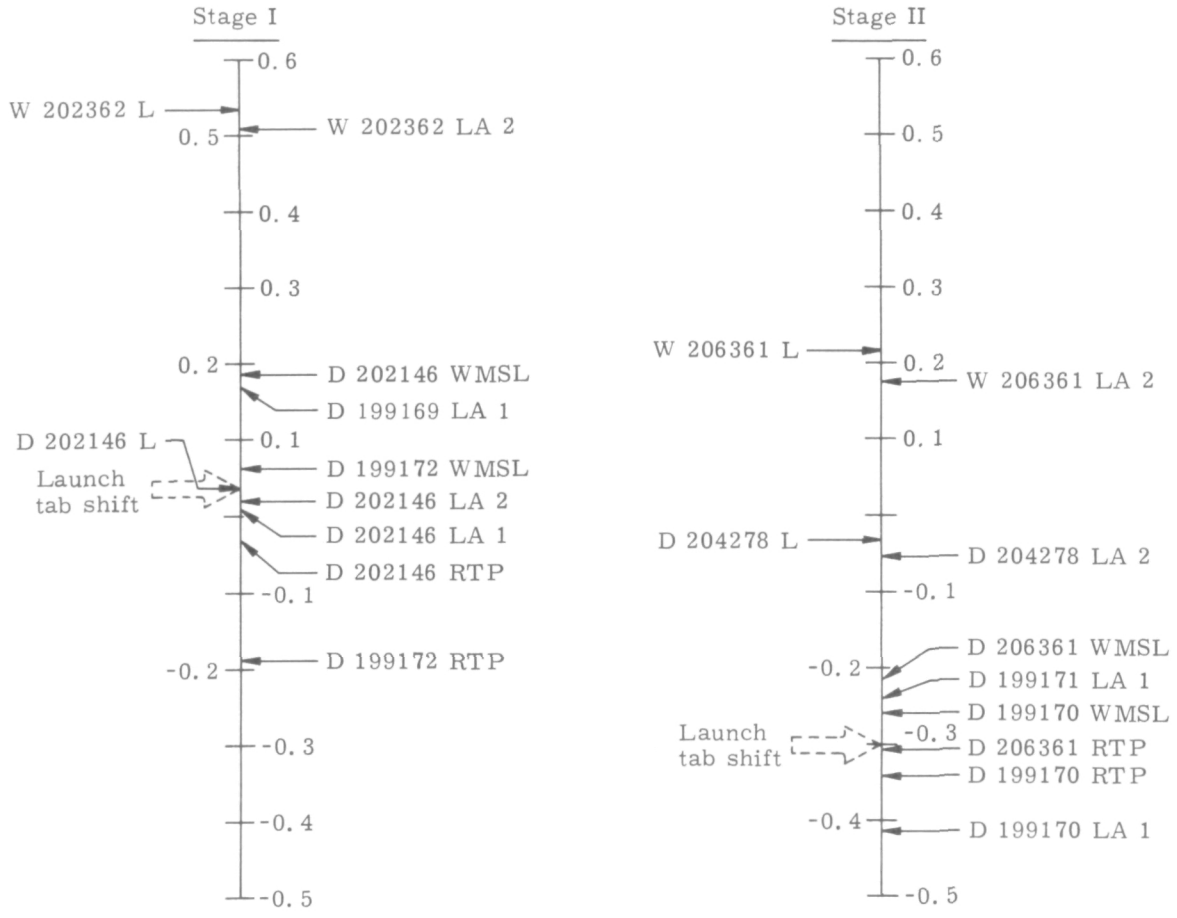
Data are corrected for flow rate and meter verification results, where available. All data are referenced to original tank calibration and represent the percent error of the flowmeter result at hi-lite from the original calibration.



LEGEND

D	= Denver
W	= Wyle
RTP	= first loading
WMSL	= wet mock
LA 1	= 25 October 1965 launch attempt
LA 2	= 12 December 1965 launch attempt
L	= launch

Fig. III-15. GLV-6 Loading Summary--Fuel

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TABLE III-15

Verification of Mission Loads

Tank	Flowmeter Indicated Load (lb)	Requested Load (lb)	Flight Verification Load (lb)	Difference Between Flight Verification and Flowmeter Load (%)
Stage I oxidizer	172,799*	172,789	172,780	-0.011
Stage II oxidizer	38,141*	38,141	38,256	+0.302
Stage I fuel	90,381	90,374	90,349	-0.035
Stage II fuel	21,998	21,992	21,952	-0.209

*Mission load obtained by K-factor ratio technique.

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c. Propellant assay

Prelaunch data from the propellant assay report (sampled on 6 December 1965) for oxidizer and fuel are presented in Table III-16. Specification values are also listed. Good agreement was obtained between the analysis and specification requirements. Data are from the primary RSV propellants which were used to load the vehicle.

TABLE III-16

Propellant Assay Summary

Fuel MIL-P-27402 (USAF)	Test	Requirement
Hydrazine	51.7%	51 ± 0.9%
UDMH	47.9%	46.9% min
H_2O	0.4%	2.0% max
Total N_2H_4 + UDMH	99.6%	98% min
Solids	0.2 mg/liter	25 mg/liter
Particles on 50 mesh screen	0	0
Density (gm/cc) at 77° F	0.8999	--
Oxidizer MIL-P-26539 (USAF)	**Test**	**Requirement**
Nitrogen tetroxide (N_2O_4)	99.8%	99.4% min
Chloride as NOCl	*	--
H_2O equivalent	0.01%	0.2%
Solids	0	10 mg/liter
Nonvolatile ash	*	--
Particles on 50 mesh screen	0	0

*Not reported.

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2. Propellant Temperature

a. Weather

A comparison of the F-45 day prediction, the F-1 day prediction and the actual weather for the 15 December launch of GT-6 is presented in Table III-17. The F-45 day prediction was based on weather for a hot December through March day. There was, in general, better agreement between the F-45 day prediction and actual weather than between between the F-1 day prediction and actual.

Predicted wind speed average was approximately 52% higher than actual.

TABLE III-17

Predicted and Actual Weather Conditions for GT-6 Launch

Time (est)	Dry Bulb Temperature (°F)			Dew Point Temperature (°F)			Wind Speed (kn)			Cloud Cover		
	F-45	F-1	Actual	F-45	F-1	Actual	F-45	F-1	Actual	F-45	F-1	Actual
2100 14 Dec	71.2	67.0	68.0	65.8	61.0	66.0	7	7	3	0.5	0.2	0.8
2200	70.7		68.2	65.3		66.0	7		1	0.5		0.6
2300	70.1		68.4	64.9		66.0	7		4	0.5		1.0
0000 15 Dec 1965	69.8	63.0	68.8	64.6	58.0	66.0	7	6	4	0.4	0.2	1.0
0100	69.5		68.5	64.4		66.0	7		7	0.5		0.8
0200	69.3		69.0	64.2		66.0	7		4	0.5		0.8
0300	69.0	59.0	69.3	64.0	53.0	66.0	7	6	6	0.5	0.2	0.6
0400	68.8		70.5	63.9		66.0	7		7	0.5		0.8
0500	68.7		69.2	63.7		66.0	7		7	0.5		0.8
0600	68.5	59.0	68.4	63.6	55.0	65.0	7	6	6	0.5	0.3	0.6
0700	68.4		67.3	64.9		64.0	7		5	0.6		0.6
0800	70.9		67.3	66.1		64.0	7		6	0.6		0.5
0900	72.8	63.0	70.1	67.2	57.0	66.0	9	9	7	0.6	0.4	0.4
1000	75.0			66.2			10			0.6		
1100	76.4			68.8			12			0.6		
1200	77.2	68.0		69.2	60.0		12	13		0.6	0.4	

b. Propellant loading temperatures

Table III-18 compares the requested propellant temperatures at the RSV (at start of loading) and the tank bottom probe (at hi-lite) with the measured propellant temperatures.

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TABLE III-18

Propellant Temperature Comparison--
RSV and Tank Bottom Probe

System	RSV Temperature (° F)			Tank Bottom Probe Temperature (° F)		
	Requested	Actual	Δ	Requested	Actual	Δ
Stage I fuel	26.7	26.5	0.2	28.4	28.2	-0.2
Stage II fuel	26.7	26.5	0.2	28.5	28.5	-1.0
Stage I oxidizer	25.5	25.5	0	27.9	29.8	+1.9
Stage II oxidizer	25.5	25.5	0	29.7	28.8	-0.9

The requested oxidizer RSV temperatures were matched exactly, and the fuel RSV and fuel and oxidizer tank bottom probe readings were within an acceptable range of accuracy.

RSV and flowmeter temperatures recorded during loading are shown in Figs. III-16 and III-17.

c. Liftoff temperatures

A comparison of predicted, actual and reconstructed propellant bulk temperatures appears in Table III-19.

TABLE III-19

Propellant Bulk Temperature Comparison

System	F-45 Day Prediction (° F)	F-1 Day Prediction (° F)	Actual (° F)	Reconstructed (° F)
Stage I fuel	38.7	38.1	41.1	40.8
Stage II fuel	37.8	38.2	41.8	39.2
Stage I oxidizer	39.8	39.8	42.0	41.6
Stage II oxidizer	42.4	43.6	44.2	44.0

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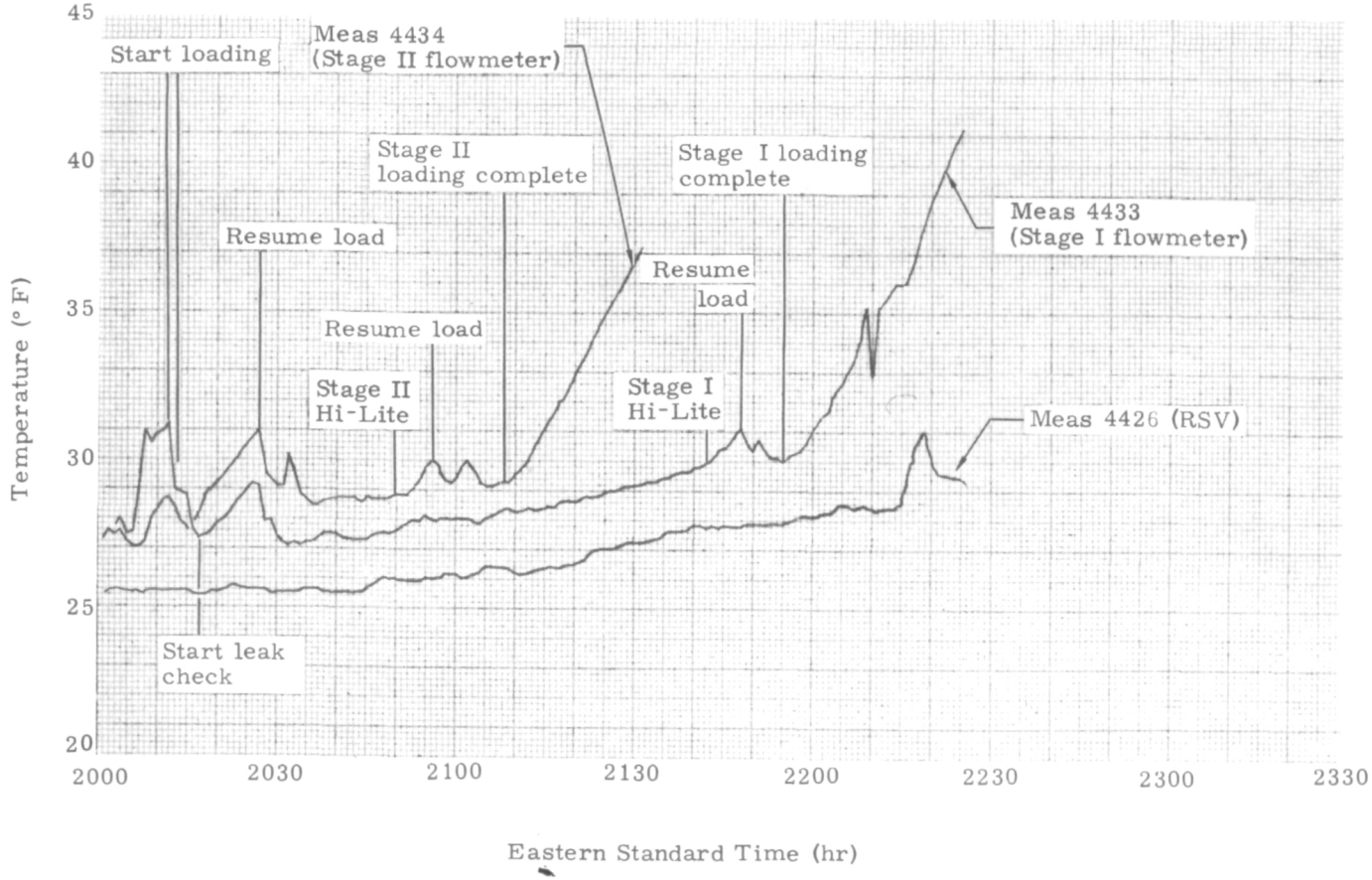

Fig. III-16. Oxidizer Temperatures During Loading

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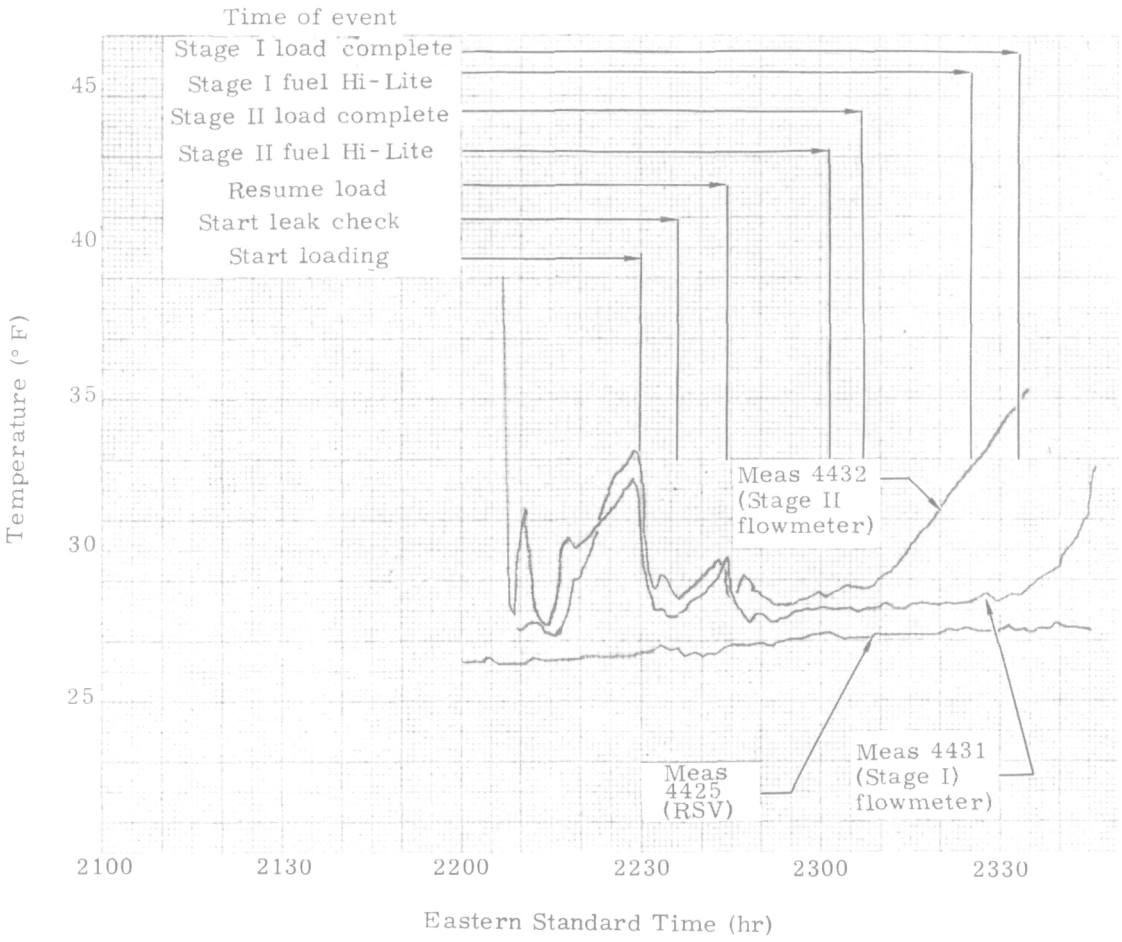

Fig. III-17. Fuel Temperature During Loading

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Actual bulk temperatures at liftoff were obtained from a computer program analysis of flight data. The positions of the reconstructed, actual and predicted temperatures in the mixture ratio band are shown in Figs. III-18 and III-19.

Figures III-20 through III-23 show a comparison of the F-1 day temperature prediction, the reconstructed temperature and actual propellant probe temperatures during the countdown for each propellant tank.

Correlation of actual and reconstructed temperatures was excellent. The difference between the F-1 day prediction was due to the differences between predicted and actual weather conditions. Average actual dry bulb and wet bulb temperatures were 5.5° F and 8.3° F higher, respectively, than predicted. Predicted winds averaged 3 knots higher.

The maximum deviation between actual and reconstructed temperatures (10.3% of total rise) occurred in the Stage II fuel tank. All others were 3.8% or less.

d. Suction temperatures

The actual pump inlet temperatures were in good agreement with the predicted temperature profiles. These data are shown in Figs. III-24 through III-27. The trends of the actual temperature curves were in good agreement with those predicted. Deviations may be ascribed to differences in predicted and actual weather and the differences between optimum and T-0 temperatures. In Table III-20 a comparison is made between the suction and tank bottom probes at various times after FS_1.

TABLE III-20

Propellant Temperature Comparison--Tank Bottom Probe and Pump Inlet

System	Time (sec)	Suction Probe Temperature (° F)	Tank Bottom Temperature Probe (° F)	Delta Temperature (° F)
Stage I fuel	FS_1 + 5	40.2	40.9	-0.7
Stage II fuel	FS_1 + 25	39.1	38.3	+1.4
Stage I oxidizer	FS_1 + 6	39.9	39.0	+0.9
Stage II oxidizer	FS_1 + 22	42.8	43.3	-0.5

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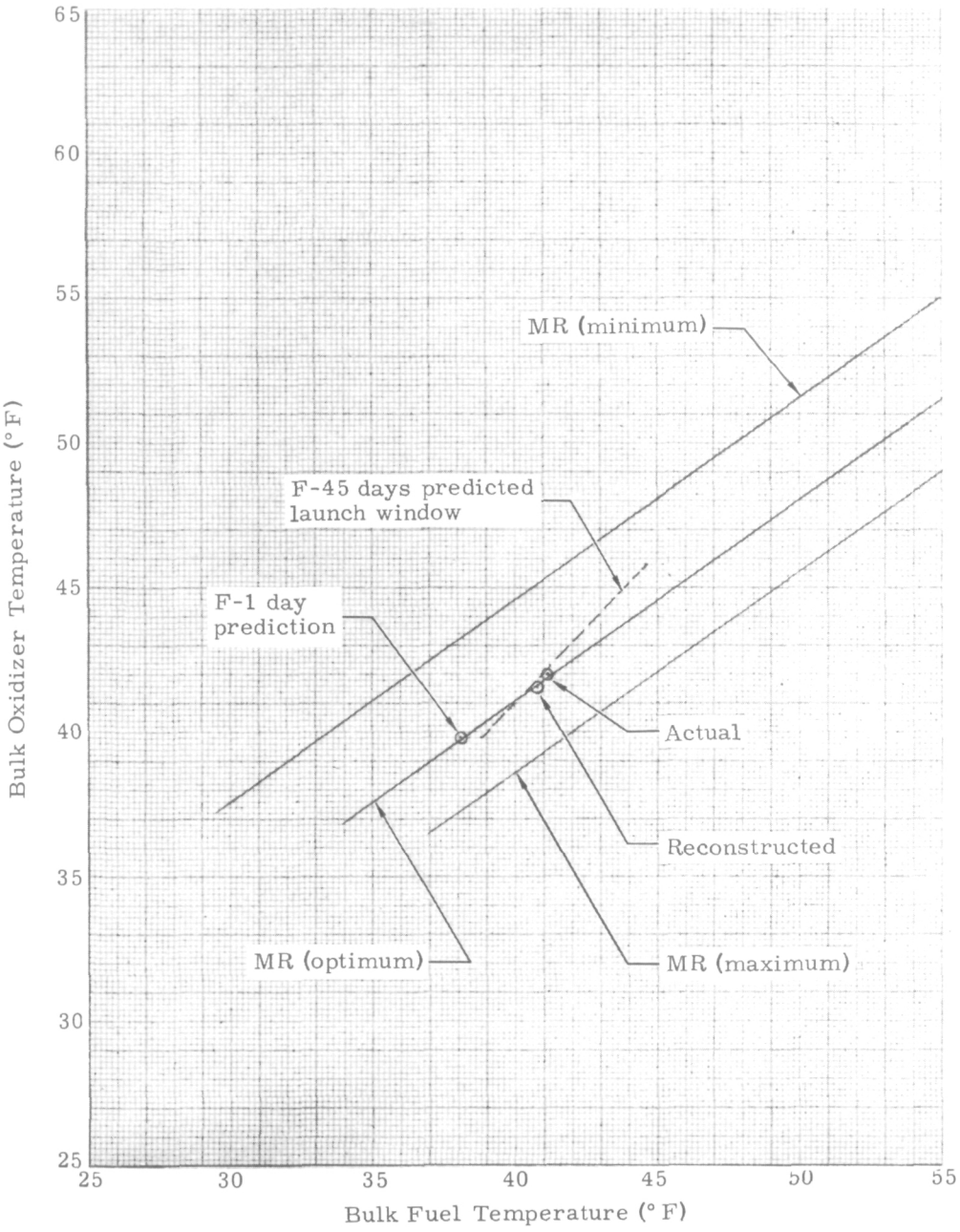

Fig. III-18. Propellant Bulk Temperatures at Liftoff, Stage I

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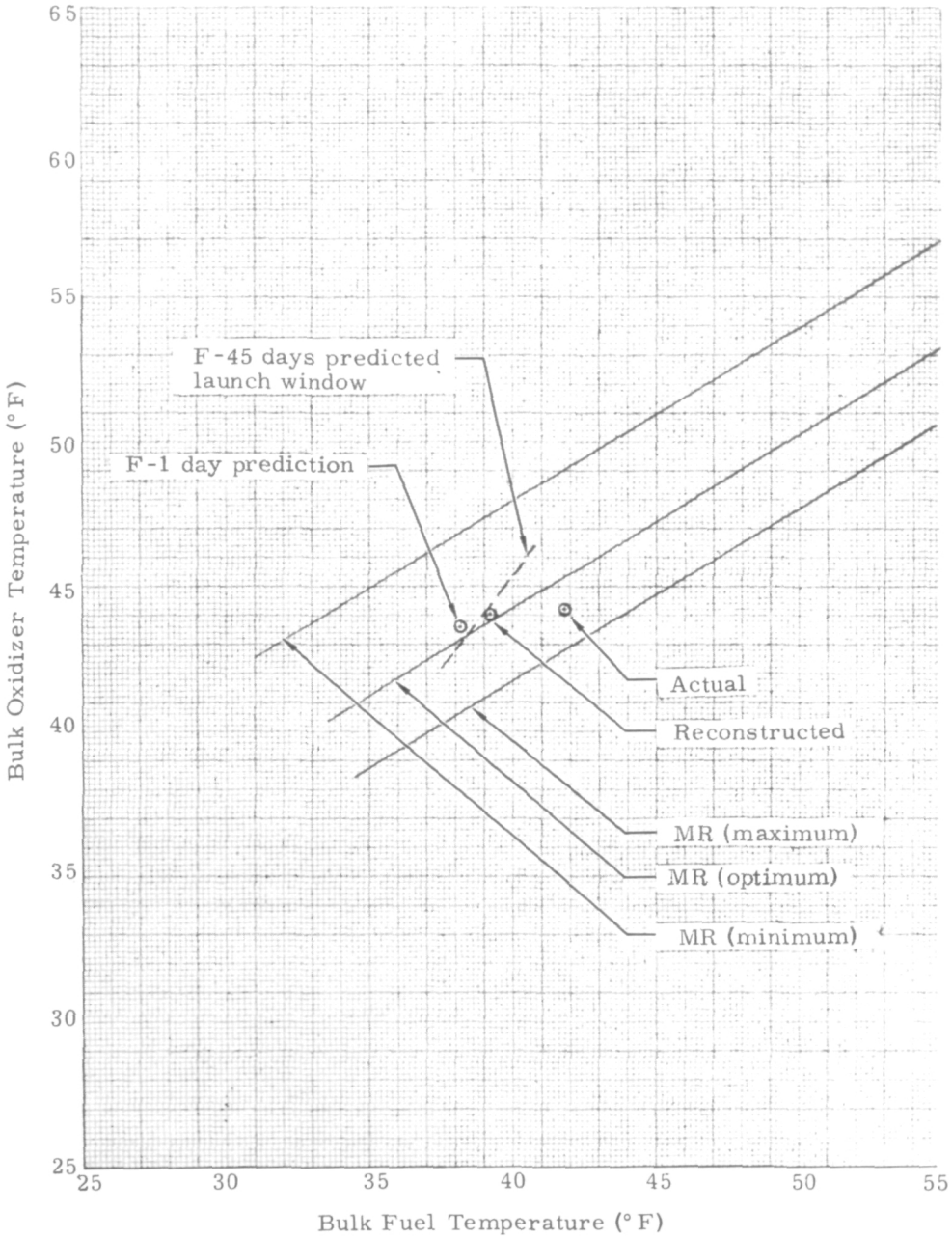

Fig. III-19. Propellant Bulk Temperatures at Liftoff, Stage II

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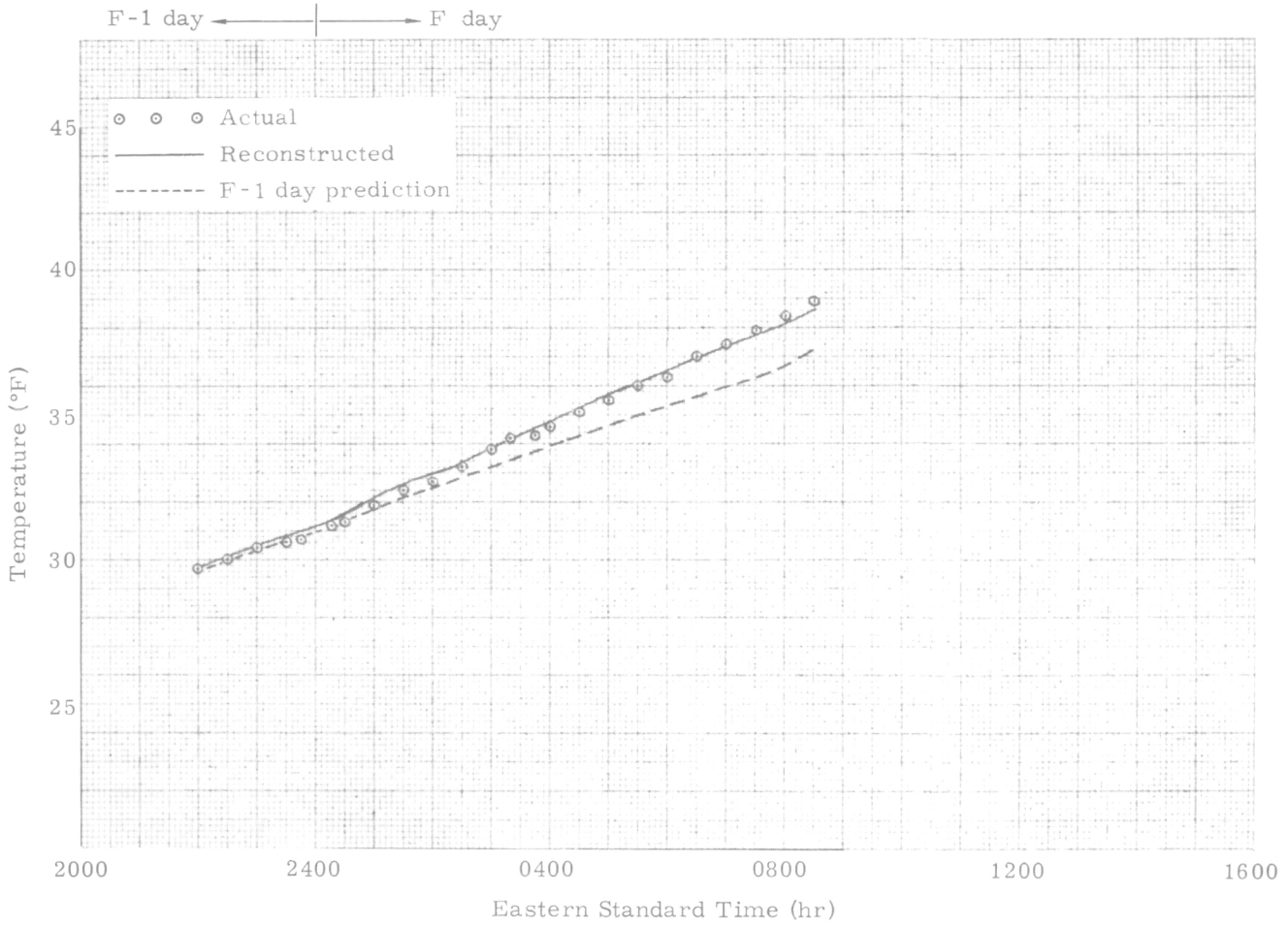

Fig. III-20. Stage I Oxidizer Tank Bottom Probe Temperature (Meas 4128)

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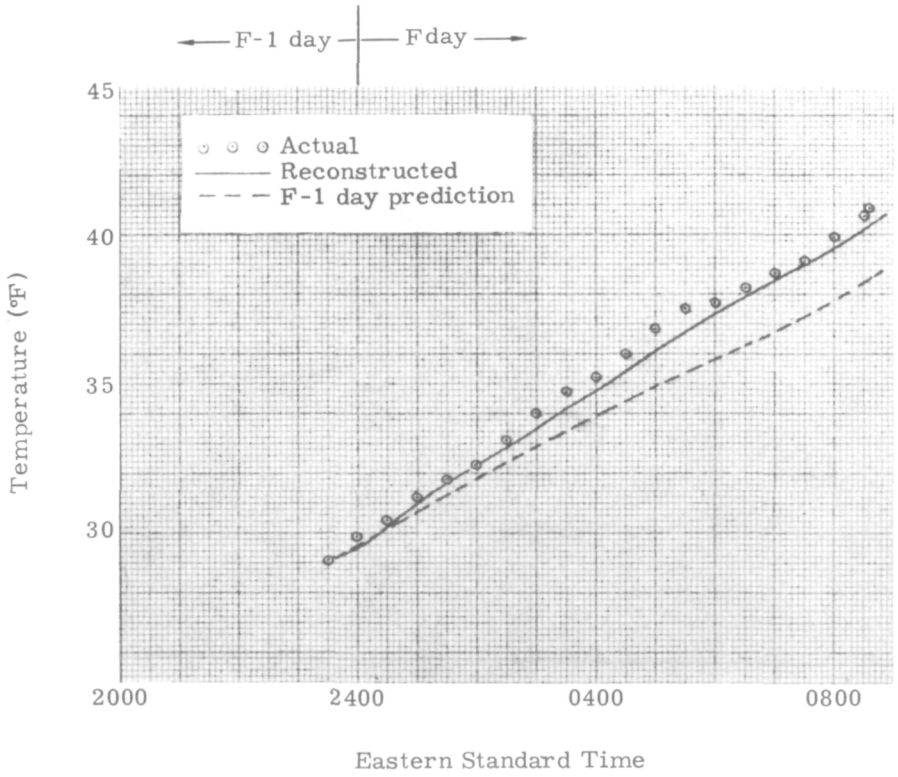

Fig. III-21. Stage I Fuel Tank Bottom Probe Temperature (Meas 4124)

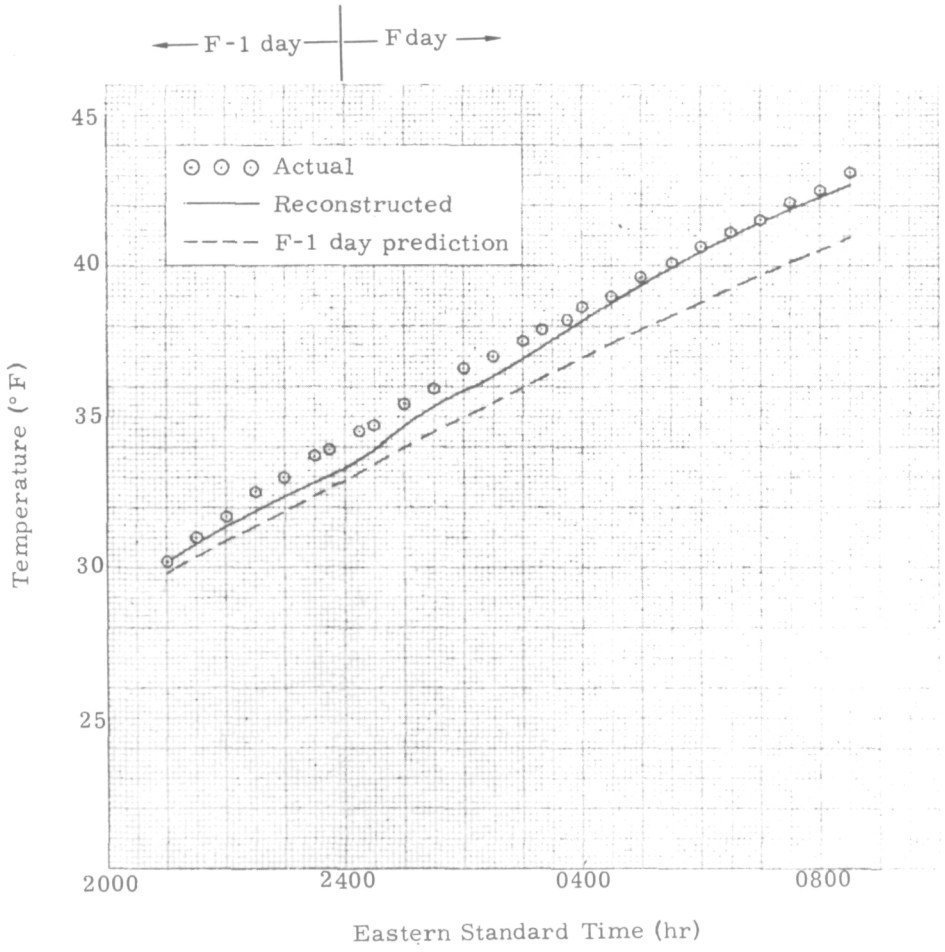

Fig. III-22. Stage II Oxidizer Tank Bottom Probe Temperature (Meas 4604)

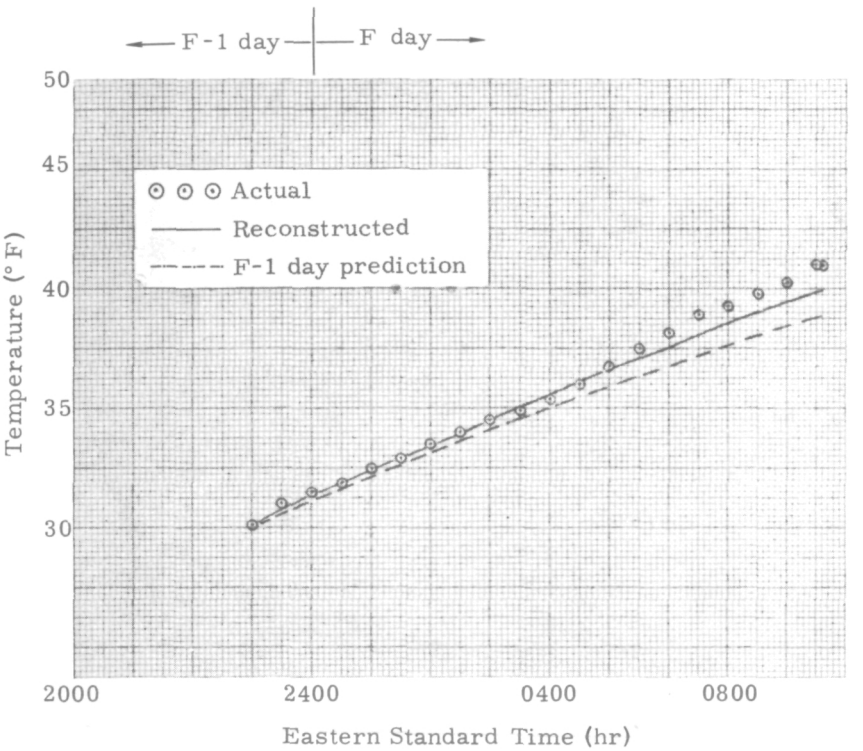

Fig. III-23. Stage II Fuel Tank Bottom Probe Temperature (Meas 4601)

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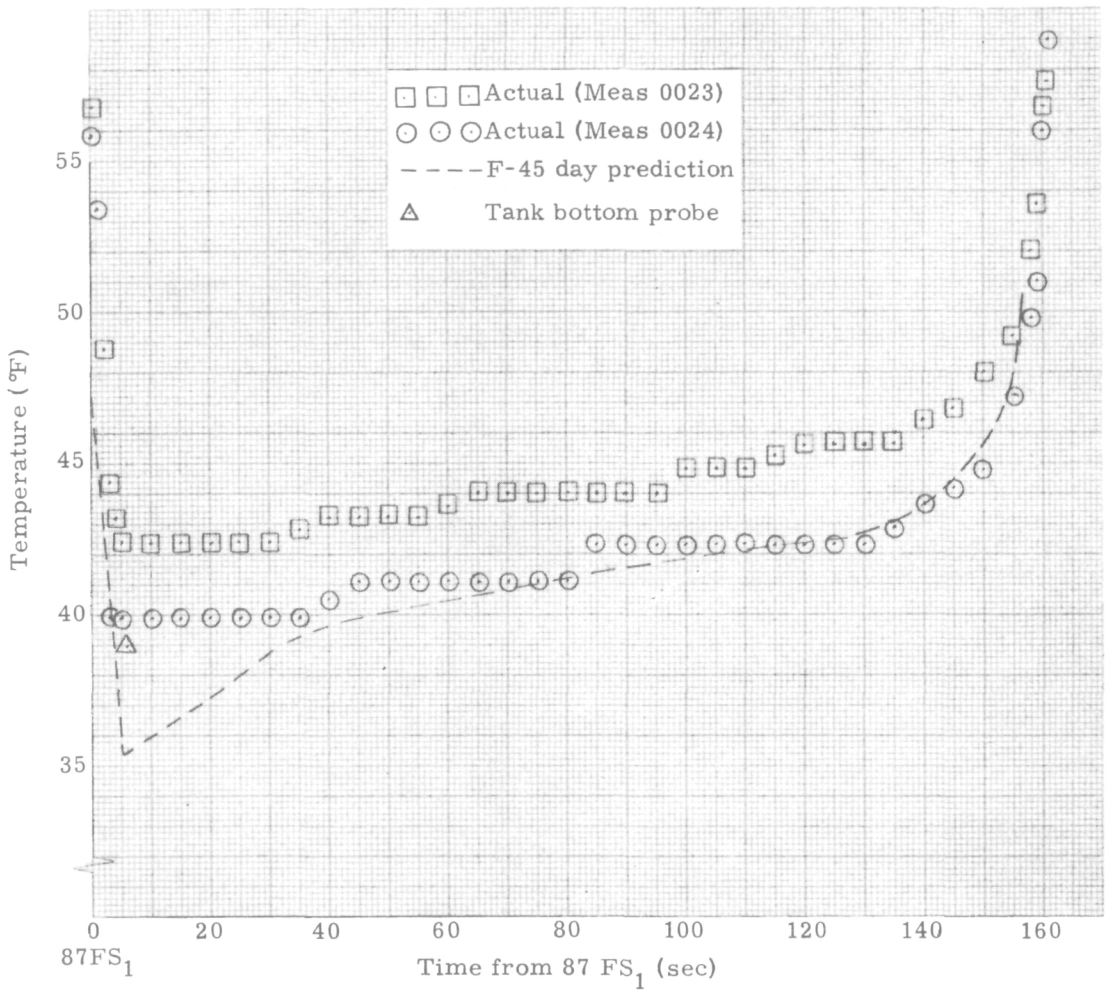

Fig. III-24. Stage I Oxidizer Pump Inlet Temperature (Meas 0023 and 0024)

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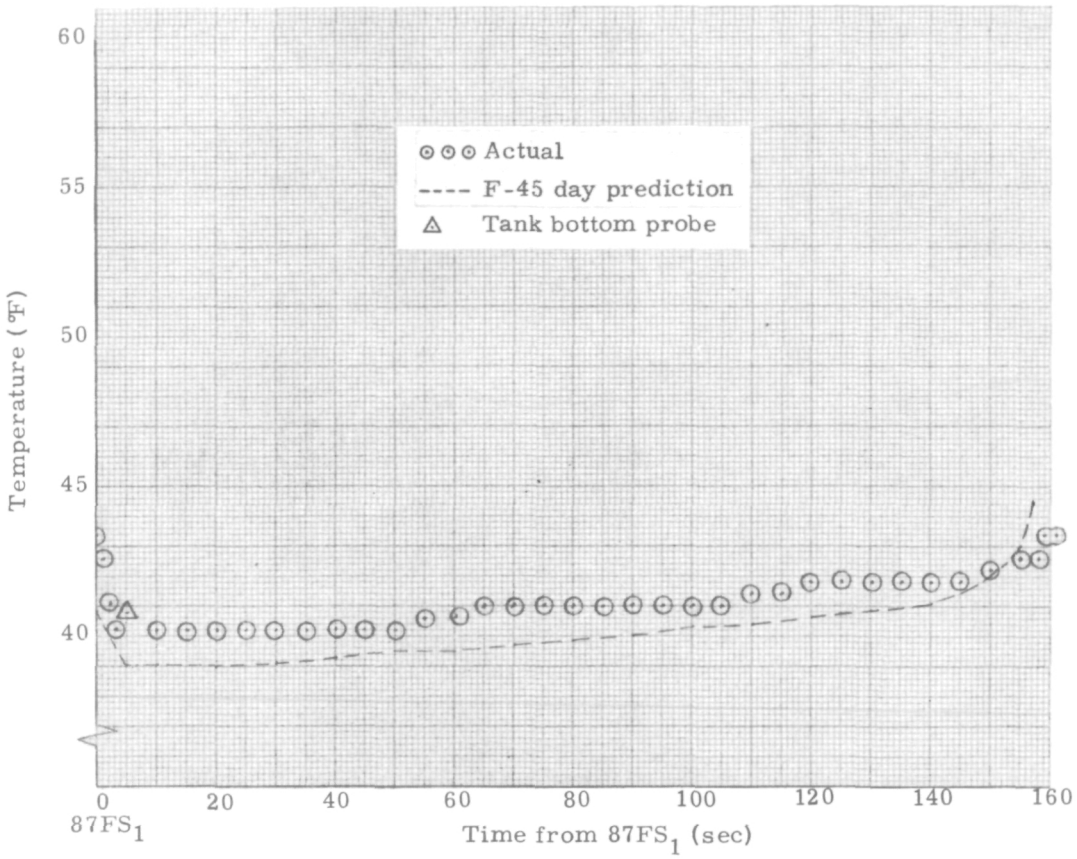

Fig. III-25. Stage I Fuel Pump Inlet Temperature (Meas 0013)

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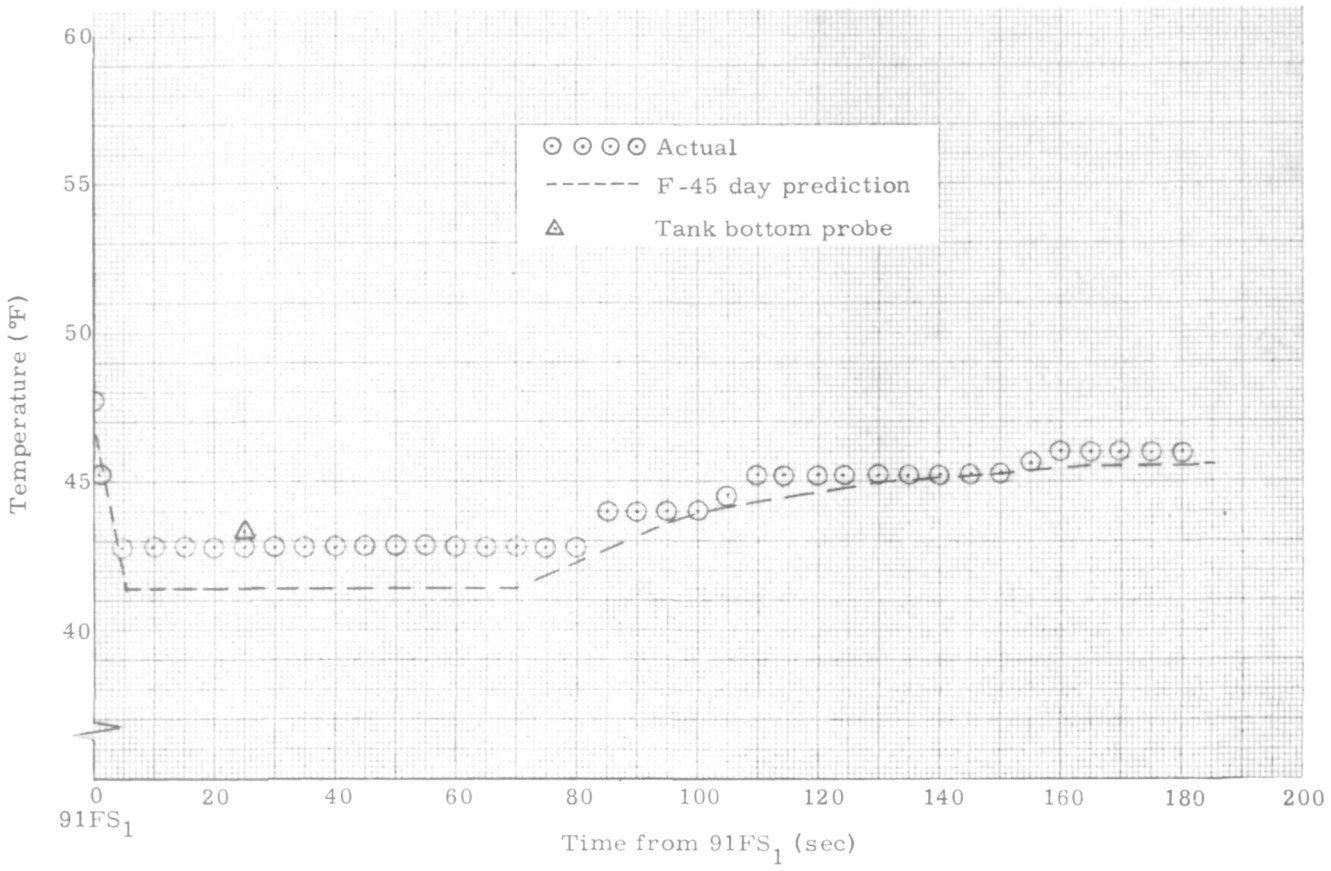

Fig. III-26. Stage II Oxidizer Pump Inlet Temperature (Meas 0514)

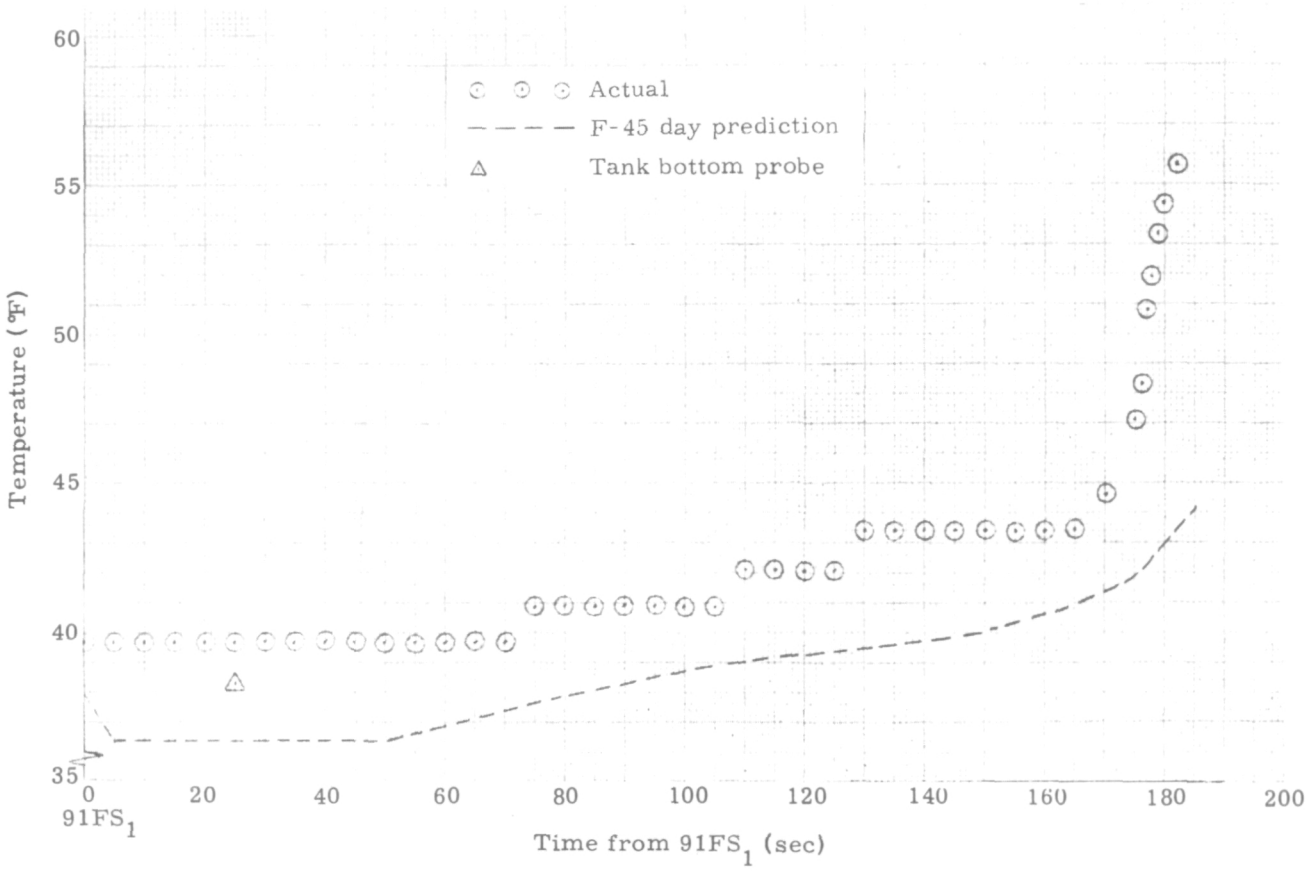

Fig. III-27. Stage II Fuel Pump Inlet Temperature (Meas 0508)

3. Propellant Feed System

a. Feedline transients

The maximum transient pressures recorded at the pump inlet instrumentation bosses are listed in Table III-21.

TABLE III-21

Maximum Transient Pressures at Pump Inlet

Measurement	At Prevalve Opening	At Initial Pressure Wave	At Ignition (psia)	At TCV Closing (psia)	Design Operating Pressure (psia)
Stage I oxidizer (0017)	No data	Negligible	124	Negligible	215
Stage I fuel (0014)	No data	Negligible	Negligible	Negligible	55
Stage II oxidizer (0510)	No data	Negligible	*	68	260
Stage II fuel (0507)	No data	Negligible	*	Negligible	80

*Not available due to telemetry staging blackout.

No data were available on the prevalve opening pressure transients, since these valves were opened during the 12 December 1965 launch attempt and were not replaced. Ignition transient pressures were, in general, similar to those of GLV-5 and GLV-7 flights. Telemetry blackout normally experienced during Stage II ignition eliminates data on sustainer engine ignition transients.

b. Pump inlet suction pressures

Stage I and Stage II static suction pressures at the suction measurement boss locations are shown in Figs. III-28 through III-31, which present the preflight predicted, postflight reconstructed and best estimate of actual flight pressures. The postflight reconstructed curves were based on flight measured values of ullage gas pressure, axial load factors, propellant temperatures and propellant loadings.

The Stage I oxidizer best estimate curve of the static suction pressures at the measurement boss (Meas 0017) consists of an average of the measured pressure and the two oxidizer standpipe pressures

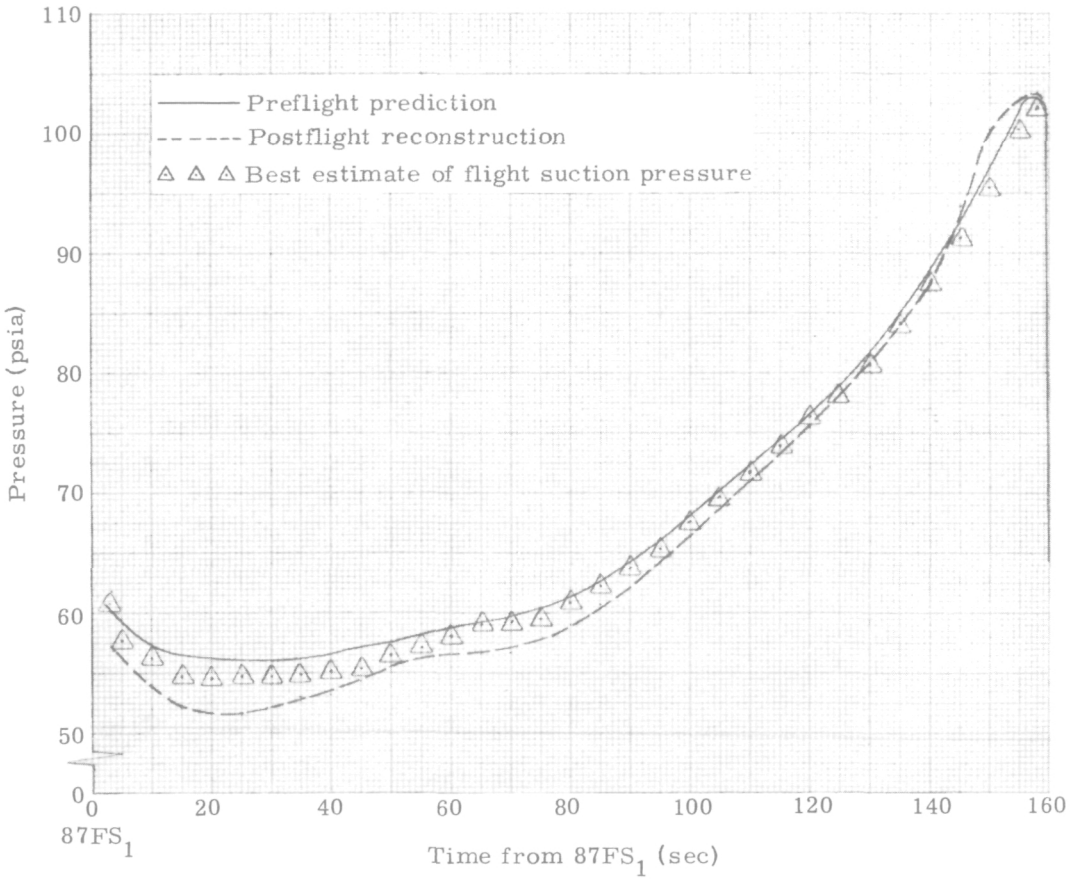

Fig. III-28. Stage I Oxidizer Suction Pressure (Meas 0017)

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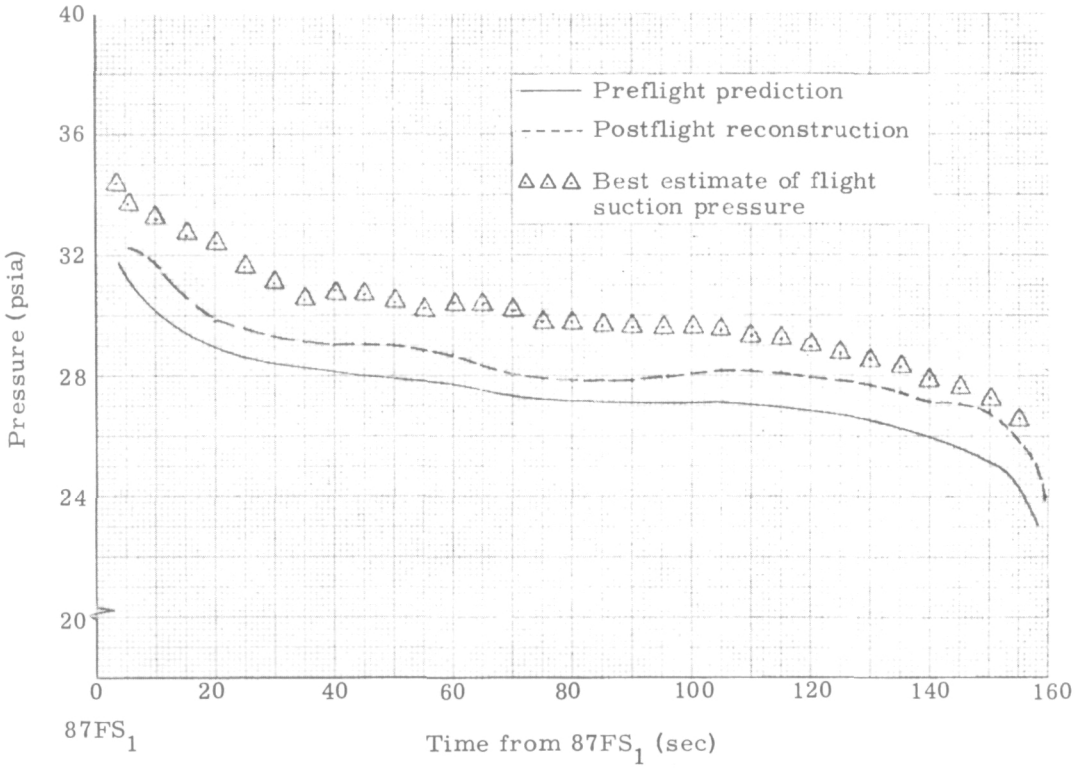

Fig. III-29. Stage I Fuel Suction Pressure (Meas 0014)

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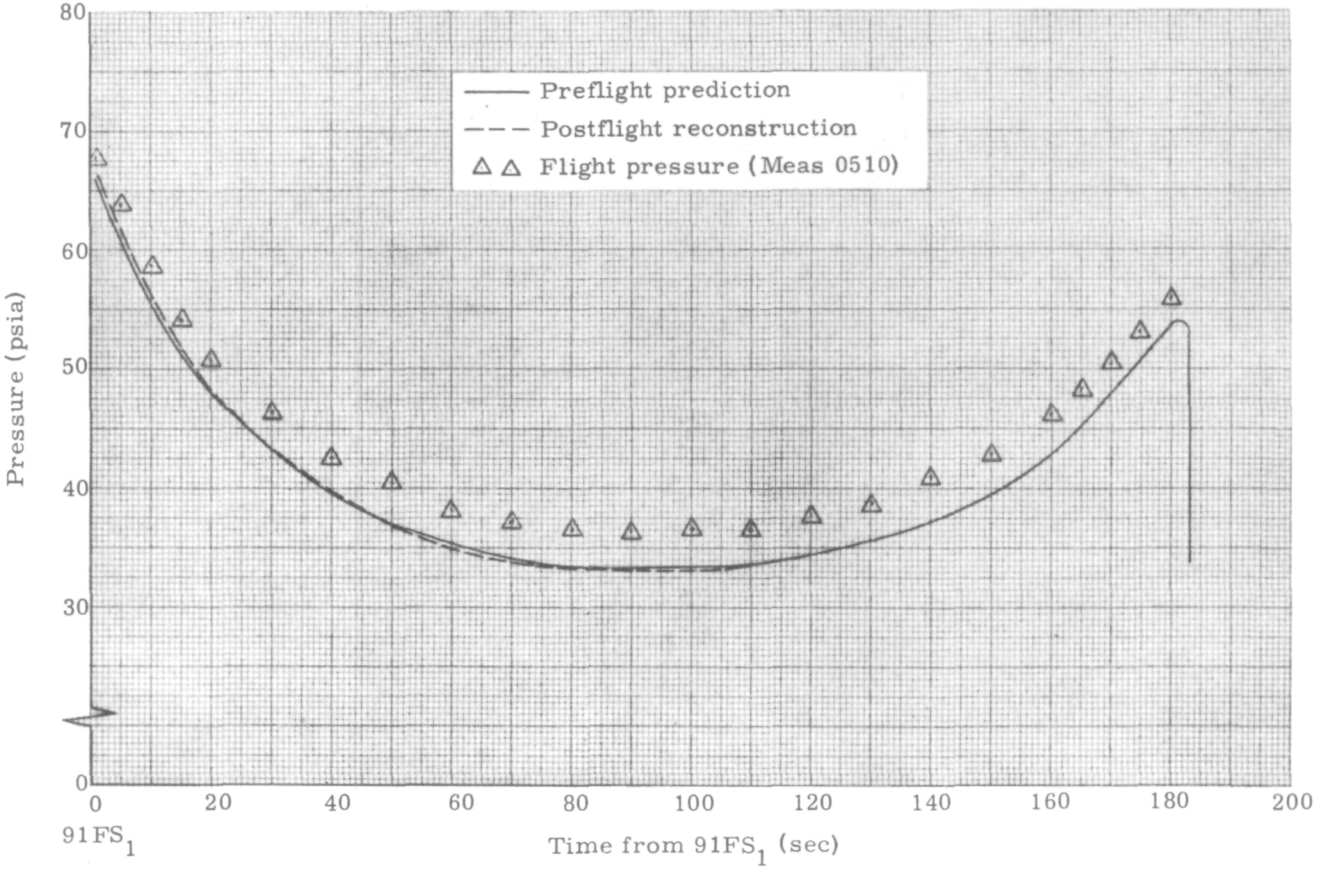

Fig. III-30. Stage II Oxidizer Suction Pressure (Meas 0510)

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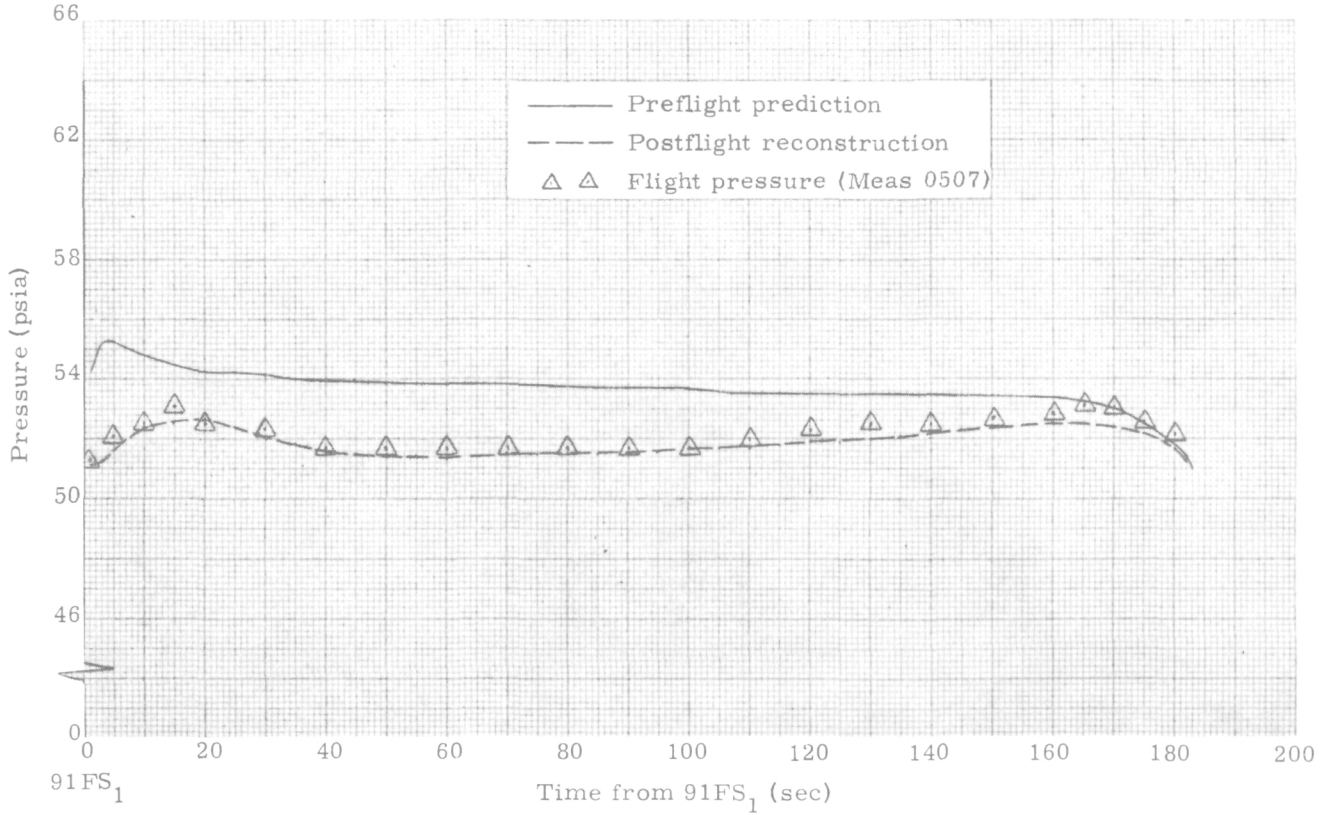

Fig. III-31. Stage II Fuel Suction Pressure (Meas 0507)

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(Meas 0033 and 0034) adjusted to the Meas 0017 boss location. The Stage I fuel suction pressure best estimate at Meas 0014 boss location is an average of measured pressure and the two fuel accumulator pressures (Meas 0037 and 0038) adjusted to the Meas 0014 boss location from $87FS_1$ to $87FS_1$ + 70 seconds. After this time, the best estimate of suction pressure is made up of the average of Meas 0014 and Meas 0037 adjusted to the Meas 0014 location. The Stage II oxidizer and fuel best estimate suction pressures are the pressures measured by Meas 0510 and 0507, respectively.

The Meas 0038 S/A 2 fuel accumulator pressure transducer indicated an erroneous pressure after $87FS_1$ + 70 seconds. At this time the indicated pressure increased slowly until, by $87FS_1$ + 140 seconds, it was 7 psi higher than Meas 0037. Fuel accumulator piston displacement was similar on both subassemblies, indicating approximately equal pressures at the accumulator locations.

c. *NPSH supplied*

The NPSH supplied at the engine turbopump inlets during the start phase and during steady-state operation is shown in Table III-22.

4. Propellant Utilization

a. Level sensor uncover

Figures III-32 and III-33 show the predicted, actual and reconstructed level sensor uncover times for Stages I and II. Measured level sensor uncover times are tabulated in Table III-23.

Slosh, as indicated by on and off signals at the time of level sensor uncovering, was minimal on this flight. Except for the Stage II fuel high level sensor, all sensor uncoverings were clean.

b. Best estimate level sensor uncover times

Table III-24 contains the best estimate average level sensor uncover times for the GT-6 flight. Also shown are the integrated average temperatures between level sensor uncoverings and the corresponding densities. The measured average uncover times shown in Table III-13 were decreased by 0.058 second to allow for the built-in level sensor delay of 0.033 second and for the PCM digital sampling rate of 0.05 second by adding 0.025 second.

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TABLE III-22

Minimum NPSH Supplied

System	Minimum NPSH Supplied During Start Transient (psia)	Starting NPSH Required ① (psia)	Limited Operation NPSH Required ②	Minimum Steady State NPSH Supplied (psia)	Unlimited Operation NPSH Required ① (psia)
Stage I oxidizer	20.3	④	28.2 psia for 135 sec	51.5	46.1
Stage I fuel	29.0	17.1	17.1 psia for 120 sec	30.9	23.1
Stage II oxidizer	③	38.4	19.2 psia for 120 sec	32.3	22.4
Stage II fuel	③	39.8	39.8 psia for 120 sec	48.5	43.4

① Extracted from SCNP S75015, page 1, and SCNP S85008, page 1. Oxidizer values represent NPSH required at FS_1 and fuel values at engine start.

② Time given in table represents maximum cumulative operating time from engine acceptance at the given NPSH levels. The data were extracted from source ①.

③ Not available due to telemetry blackout.

④ 38.4 psia required at FS_1. No specific requirement during start transient.

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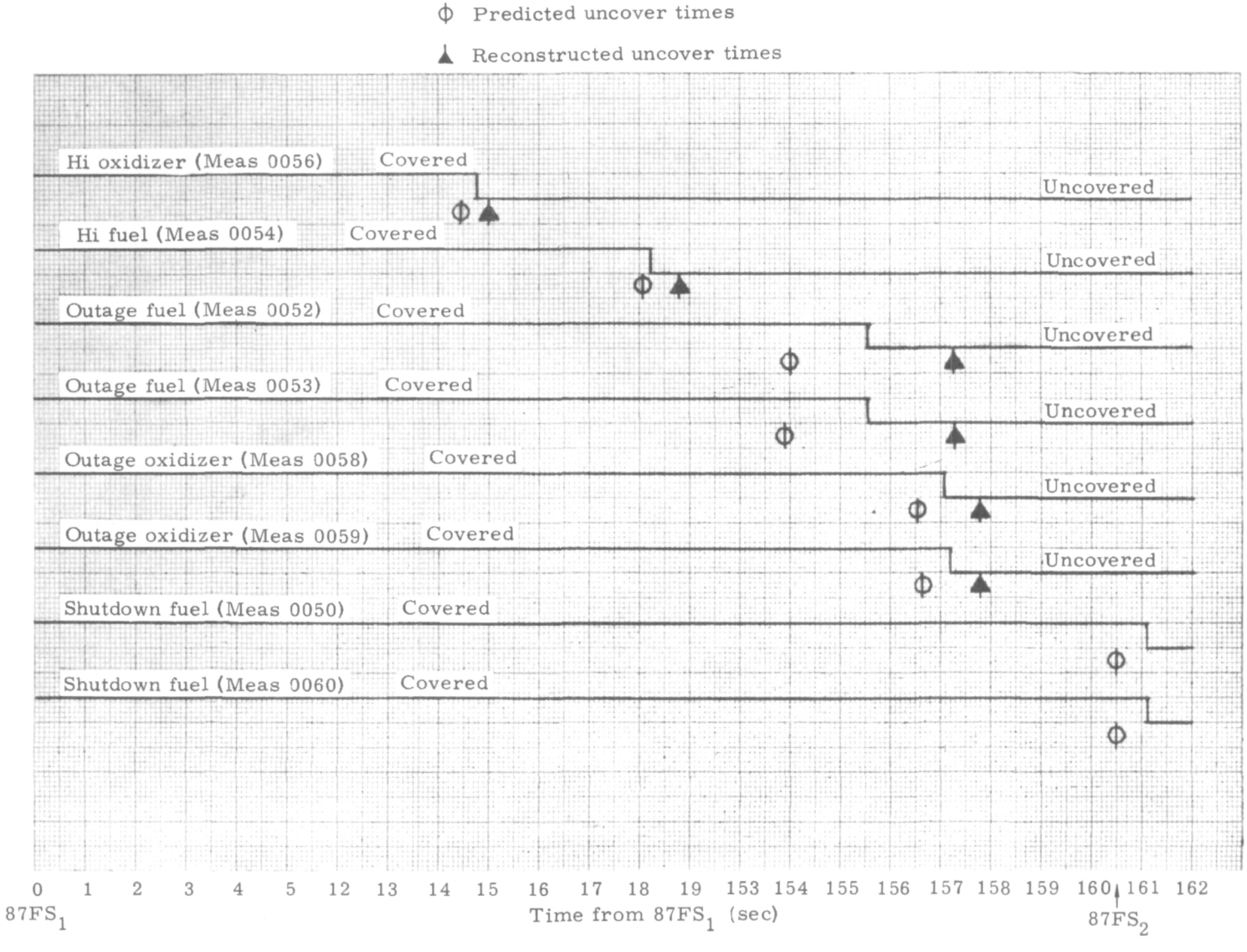

Fig. III-32. Stage I Level Sensor Operation

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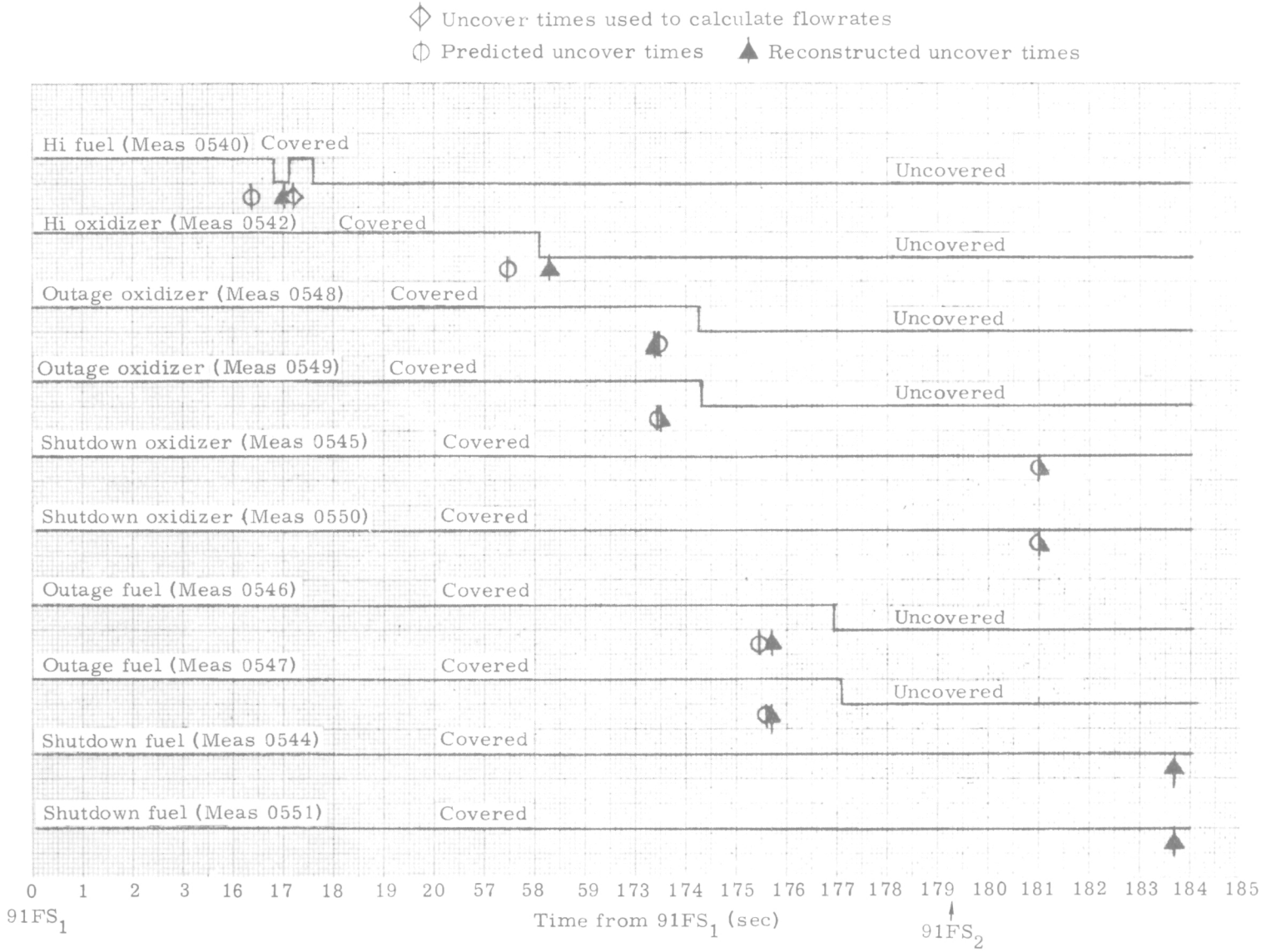

Fig. III-33. Stage II Level Sensor Operation

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TABLE III-23

Measured Level Sensor Uncover Times

Meas	Sensor	Uncover Time (GMT)	Average Uncover Time (GMT)	t (sec)	Remarks
0056	Stage I oxidizer high	1337:38.032		Refer to Table III-24	Clean
0058	Stage I oxidizer outage	1340:00.337	1340:00.387		Clean
0059	Stage I oxidizer outage	1340:00.437			Clean
0054	Stage I fuel high	1337:41.482			Clean
0052	Stage I fuel outage	1339:58.817	1339:58.797		Clean
0053	Stage I fuel outage	1338:58.777			Clean
0050	Stage I fuel shutdown				Did not uncover
0060	Stage I fuel shutdown				Did not uncover
0542	Stage II oxidizer high	1341:01.785			Clean
0548	Stage II oxidizer outage	1342:57.942	1342:57.967		Clean
0549	Stage II oxidizer outage	1342:57.992			Clean
0540	Stage II fuel high	1340:20.911			0.75 sec slosh, recover 40 sec after uncover
0546	Stage II fuel outage	1343:00.632	1343:00.707		Clean
0547	Stage II fuel outage	1343:00.782			Clean
0545	Stage II oxidizer shutdown				Did not uncover
0550	Stage II oxidizer shutdown				Did not uncover
0544	Stage II fuel shutdown				Did not uncover
0551	Stage II fuel shutdown				Did not uncover

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TABLE III-24

Best Estimate Average Level Sensor Uncover Time

Quadrant	Meas	Sensor	Average Uncover Time (GMT)	Δt (sec)
I	0056	Stage I oxidizer high	1337:37.974	442.355
I/IV	0058	Stage I oxidizer outage	1340:00.329	
II/III	0059	Stage I oxidizer outage		
I	0054	Stage I fuel high	1337:41.424	137.315
III	0052	Stage I fuel outage	1339:58.739	
II	0053	Stage I fuel outage		
I	0050	Stage I fuel shutdown		
III	0060	Stage I fuel shutdown		
IV	0542	Stage II oxidizer high	1341:01.727	116.182
I/IV	0548	Stage II oxidizer outage	1342:57.909	
II/III	0549	Stage II oxidizer outage		
I	0540	Stage II fuel high	1340:20.853	159.796
I	0546	Stage II fuel outage	1343:00.649	
III	0547	Stage II fuel outage		
II	0545	Stage II oxidizer shutdown		
IV	0550	Stage II oxidizer shutdown		
II	0544	Stage II fuel shutdown		
IV	0551	Stage II fuel shutdown		

System	Integrated Average Temperature (between uncoverings)	Corresponding Density (lb/ft^3)
Stage I oxidizer	42.0	92.234
Stage I fuel	41.1	57.252
Stage II oxidizer	44.7	92.025
Stage II fuel	41.7	57.234

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Table III-25 lists the level sensor volumes and delta volumes used in the level sensor flow rate analysis. The Stages I and II oxidizer and fuel high-level sensor volumes were reconstructed to reflect the volumes which were determined by calibration at Cape Kennedy using the propellant transfer and pressurization system. The Stages I and II outages and shutdown level sensor volumes were calculated using the actual counts of flowmeter pulses obtained during the special loading and the WMSL exercises.

TABLE III-25

Averaged Volumes at Level Sensor Locations

Tank	Sensor	Averaged Volumes (stretch included) (ft^3)	Volume (ft^3)
Stage I oxidizer	Hi-level	1708.20	1670.35
	Outage	37.85	
Stage I fuel	Hi-level	1402.54	1336.74
	Outage	65.80	
Stage II oxidizer	Hi-level	285.51	263.10
	Shutdown	22.41	
Stage II fuel	Hi-level	350.08	331.00
	Outage	19.08	

c. Flow rates

Table III-26 presents the predicted and the actual volumetric flow rates between level sensors.

TABLE III-26

Propellant Volumetric Flow Rate

Tank	Predicted (ft^3/sec)	Actual (ft^3/sec)
Stage I oxidizer	11.751	11.734
Stage I fuel	9.821	9.735
Stage II oxidizer	2.269	2.265
Stage II fuel	2.087	2.071

d. Mixture ratio

Table III-27 compares the Stages I and II predicted and actual engine mixture ratios (between level sensors) for the GLV-6 flight.

TABLE III-27

Engine Mixture Ratio

System	Predicted Mixture Ratio	Mixture Ratio Actual
Stage I	1.9280	1.9418
Stage II	1.7470	1.7578

Sensitivity coefficients applied to the delta between the predicted and actual variations in average suction pressure and temperature between sensor uncoverings yield the information shown in Table III-28.

TABLE III-28

Mixture Ratio Pressure and Temperature

System	Pressure (psi)	Δ Mixture Ratio (Pressure)	Temperature (°F)	Δ Mixture Ratio (temp)	Mixture Ratio (total)
Stage I oxidizer	-0.5	-0.000810	+0.9	-0.002086	-0.002514
Stage I fuel	+2.5	-0.006250	+1.0	+0.001627	-0.003028
Total Stage I		-0.007060	--	-0.000459	-0.007519
Stage II oxidizer	-1.0	-0.004200	+1.4	-0.003661	-0.006002
Stage II fuel	-1.5	+0.005310	+3.2	+0.005325	+0.008277
Total Stage II		+0.002760	--	+0.001664	+0.004424

By applying the delta mixture ratio (total) shown in Table III-28 to the predicted (F-45 Day) between-sensor mixture ratios, the run-to-run variation can be calculated. The mixture ratio deviation along with the allowable run-to-run dispersions are shown in Table III-29.

TABLE III-29

Mixture Ratio Deviation

System	Predicted Mixture Ratio (corrected for pressure and temperature variations)	Actual Mixture Ratio	Deviation (%)	Allowable Run-to-Run Dispersion (%)
Stage I	1.9205	1.9418	+1.11	±1.38
Stage II	1.7426	1.7578	+0.87	±2.28

e. Outage and trapped propellant

Table III-30 shows the mean and maximum (99%) outages predicted for GLV-6. Also shown are the actual outages as calculated using the information contained in the reconstructed propellant inventories of Tables III-36 and Table III-37.

TABLE III-30

Outage Prediction

System	Predicted (F-45 day)		Predicted (F-0 day)		Actual
	Mean	Max (99%)	Mean	Max (99%)	
Stage I	0.221%	0.645%	0.224%	0.643%	0.424% fuel
	571 lb	1669 lb	578 lb	1664 lb	834 lb
Stage II	0.343%	1.029%	0.335%	1.042%	0.323% fuel
	205 lb	614 lb	200 lb	622 lb	255 lb

All outages are presented as percent of total steady-state propellants (taken from Ref. 10) and in pounds. The value used for total steady-state propellants are 258,623 pounds for Stage I and 59,695 pounds for Stage II.

The predicted and actual trapped propellants for Stages I and II are given in Table III-31.

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TABLE III-31

Trapped Propellants

System	Oxidizer (lb)		Fuel (lb)	
	Predicted	Actual	Predicted	Actual
Stage I				
Above interface	0	0	20	20
Below interface	235	235	309	309
Stage II				
Above interface	0	0	0	0
Below interface	20	20	14	14

f. Start and holddown propellant consumptions

The predicted and actual propellant consumptions during the Stage I start and holddown period are shown in Table III-32.

TABLE III-32

Stage I Ignition and Holddown Propellant Consumptions

Stage I	Oxidizer (lb)		Fuel (lb)	
	Predicted	Actual	Predicted	Actual
Start consumption ($87FS_1$ to TCPS)	209	208	44	44
Holddown consumption (TCPS to Liftoff)	2131	2174	1127	1171

The predicted and actual start consumptions listed in Table III-32 were selected from Ref. 15 and were modified tc allow for the difference between propellant out of the tanks (as listed in the report) and propellant overboard. The predicted holddown consumption was derived from the engine analytical model and previous flight test data, whereas the actual value was derived from the Post Test Rocket Engine System Total Operation (PRESTO) engine performance reconstruction program.

The Stage II predicted and actual propellant consumptions between $91FS_1$ and $91FS_1$ + 1.2 seconds are listed in Table III-33.

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TABLE III-33

Stage II Start Propellant Consumption

	Oxidizer (lb)		Fuel (lb)	
	Predicted	Actual	Predicted	Actual
Start Consumption ($91FS_1$ to $91FS_1$ + 1.2 sec)	135	135	53	53

The predicted and actual consumptions were obtained from Ref. 15 and modified as on the Stage I start consumption.

g. Vapor retained

The predicted and actual values of vapor retained in the tanks as a result of pressurization gases and propellant vaporization during flight are shown in Table III-34.

TABLE III-34

Pressurization Gas Inventory

System	Oxidizer (lb)		Fuel (lb)	
	Predicted	Actual*	Predicted	Actual*
Stage I				
Vapor Retained:				
Oxidizer tank	325	322	0	0
Fuel tank	8	8	90	91
Vaporized	5	5	0	0
Stage II				
Pressurization Fuel tank	5	5	51	49
Vaporization Oxidizer tank	9	9	-	-

*The actual values were obtained from the reconstructed flight pressure profile of the pressurization computer program runs.

h. Shutdown

Stage I shutdown was due to oxidizer exhaustion. The predicted and actual values for the propellants consumed during shutdown are presented in Table III-35. The actual values were obtained by integrating a curve (derived from PRESTO) of flight flow rate versus time after $87FS_2$.

Stage II shutdown was initiated by a guidance command; therefore, the propellants were not exhausted as in Stage I. The predicted and actual values for the propellants consumed during shutdown are shown in Table III-35. The actual values were computed using altitude shutdown impulse data.

TABLE III-35

Propellant Shutdown Consumption

	Oxidizer (lb)		Fuel (lb)	
	Predicted	Actual	Predicted	Actual
Stage I	0	0	145	164
Stage II	78	76	62	60

i. Propellant inventory

The reconstructed propellant inventories for GT-6 are shown in Tables III-36 and III-37 for Stages I and II, respectively. The inventory consists of both nonusable and usable propellants. The burning time margin for Stage II was 2.348 seconds.

5. Components

a. Prevalves

During the launch countdown, all prevalve functions were performed without incident. Prevalves installed for the flight are identified in Table III-38.

TABLE III-36

GLV-6 Stage I Reconstructed Propellant Loading

I.	Predicted in-flight engine mixture ratio			1.9248 ± 1.54%
II.	Average in-flight mixture ratio (engine)			1.9381 ± 1.71%
III.	Outage (percent of total usable propellants)			0.32%
		Oxidizer (lb)	Fuel (lb)	Total (lb)
IV.	Nonusable propellants			
	A. Fuel bleed	0	11	11
	B. Start consumption ($87FS_1$ to TCPS)	208	44	252
	C. Holddown (TCPS to liftoff (2 sec))	2,174	1,171	3,345
	D. Trapped above interface at shutdown	0	20	20
	E. Trapped below interface at shutdown	235	309	544
	F. Vapor retained at shutdown			
	1. For pressurization			
	a. Oxidizer tank	322		322
	b. Fuel tank	8	91	99
	2. Vaporized	5		5
	G. Total nonusable	2,952	1,646	4,598
V.	Usable propellants			
	A. Steady-state overboard (liftoff to $87FS_2$)	169,828	87,705	257,533
	B. Shutdown transient (FS_2 to 0% thrust)	0	164	164
	C. Outage		834	834
	D. Total usable	169,828	88,703	258,531
VI.	Total propellant loaded	172,780	90,349	263,129
VII.	Propellant load at liftoff	170,398	89,134	259,532
VIII.	Weight of initial pressurizing gas			
	A. Fuel tank (N_2)			8
	B. Oxidizer tank ($N_2 + NO_2$)			17

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TABLE III-37

GLV-6 Stage II Reconstructed Propellant Loading

I.	Predicted in-flight engine mixture ratio			1.7448 ± 2.52%
II.	Average in-flight mixture ratio (engine)			1.7628 ± 1.55%
III.	Outage (percent of total usable propellants)			+0.424%
IV.	Burning time margin			2.348 sec
		Oxidizer (lb)	Fuel (lb)	Total (lb)
V.	Nonusable propellants			
	A. Fuel bleed	0	11	11
	B. Trapped above interface at FS_2 + 20 sec (0% thrust)	0	0	0
	C. Trapped below interface at FS_2 + 20 sec (0% thrust)	20	14	34
	D. Vapor retained after FS_2			
	1. Pressurization (fuel tank)	5	49	54
	2. Vaporization (oxidizer tank)	9		9
	E. Total nonusable	34	74	108
VI.	Usable propellants			
	A. Start consumption (FS_1 to 90% thrust)	135	53	188
	B. Steady-state overboard (90% thrust to FS_2)	37,518	21,236	58,754
	C. Shutdown consumption (FS_2 to 0% thrust)	76	60	136
	D. Steady-state residuals (after FS_2)			
	1. Burning time margin	493	274	767
	2. Outage		255	255
	E. Total usable	38,222	21,878	60,100
VII.	Total propellants loaded	38,256	21,952	60,208
VIII.	Weight of initial pressurizing gas			
	A. Fuel tank (N_2)			5
	B. Oxidizer tank (N_2 + NO_2)			32

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TABLE III-38

Prevalve Identification

Description	Part No.	Serial No.
Stage I oxidizer S/A 1 (fill and drain)	PS47510007-139	0700067
Stage I oxidizer S/A 2 (drain)	PS47510007-159	0700016
Stage II oxidizer S/A 3 (fill and drain)	PS47510005-199	0600041
Stage I fuel S/A 1 (fill and drain)	PS47510005-159	0600044
Stage I fuel S/A 2 (drain)	PS47510005-169	0600033
Stage II fuel S/A 3 (fill and drain)	PS47510006-059	0400015

b. Level sensors

GLV-6 incorporated 18 Bendix optical-type propellant level sensors. These are identified in Table III-39. All sensors performed satisfactorily during propellant loadings and in flight.

TABLE III-39

GLV-6 Propellant Level Sensor Identification

	Stage I					Stage II				
Location	Meas	Quadrant	Volume* (ft^3)	Configuration	Serial No.	Meas	Quadrant	Volume* (ft^3)	Configuration	Serial No.
Oxidizer Tank										
High level	0056	I	1708.61	-039	000241	0542	IV	284.27	-039	000326
Outage	0058	I/IV	38.39	-039	000203	0548	I/IV	21.76	-039	000239
Outage	0059	II/III	38.21	-039	000433	0549	II/III	22.88	-039	000201
Shutdown	-	-	-	-	-	0545	II	4.99	-039	000342
Shutdown	-	-	-	-	-	0550	IV	4.95	-039	000260
Fuel Tank										
High level	0054	I	1401.86	-059	000256	0540	I	350.15	-049	000248
Outage	0052	III	66.20	-049	000242	0546	I	18.01	-049	000242
Outage	0053	I	65.92	-049	000240	0547	III	18.19	-049	000229
Shutdown	0050	I	7.96	-049	000249	0544	II	1.57	-049	000228
Shutdown	0060	III	8.20	-049	000216	0551	IV	1.61	-059	000263

*Volume to interface including tank stretch at uncover time.

c. Oxidizer standpipes

The oxidizer suction line standpipes were charged at T-39 minutes during the launch attempt countdown on 12 December 1965. For the recycle operation, the airborne and AGE portions of the remote charge system were drained and purged according to procedures established as a result of the standpipe recycle problem on GT-5 (Ref. 2). The remote charge system was used at T-125 minutes during the launch countdown and no problems were encountered. Flight data obtained from pressure Meas 0033 and 0034 located in the standpipes show performance to be normal and consistent with the very low longitudinal oscillatory levels experienced on this flight.

d. Fuel accumulators

Performance of the spring-piston fuel accumulators was normal. Response of the pistons to pressure perturbations was similar to that on GLV-3, -4, -5 and -7, the primary frequency of the piston being 22 cps. As on previous flights, S/A 1 accumulator evidences a high amplitude response at approximately 5.5 cps. Figure III-34 shows the accumulator piston response from $87FS_1$ to $87FS_2$. The S/A 1 level change at $87FS_1$ + 51.5 seconds has been observed at about this same time period on all previous flights. The abrupt changes are believed to be due to time variant local pressures within Compartment 5. Local static pressures in the vicinity of the fuel accumulators are a function of the airflow entering Compartment 5 through the airscoops. No data are available to establish the static pressure variation at the accumulator vent hole; however, past analysis (Ref. 16) indicates the existence of significant velocity profiles near the accumulator.

Pressure on the feedline side of the accumulator does not vary in level at FS_1 + 51.5 seconds; consequently, any abrupt change in piston position is most likely due to a change in local Compartment 5 pressure.

Dynamic friction levels for dry accumulators were measured prior to installation of accumulator assemblies at Martin-Baltimore and again prior to flight at ETR. A summary of these friction measurements is presented in Table III-40 as peak-to-peak values (twice the equivalent friction force in one direction). Observed flight data do not indicate significant differences in friction levels between accumulators.

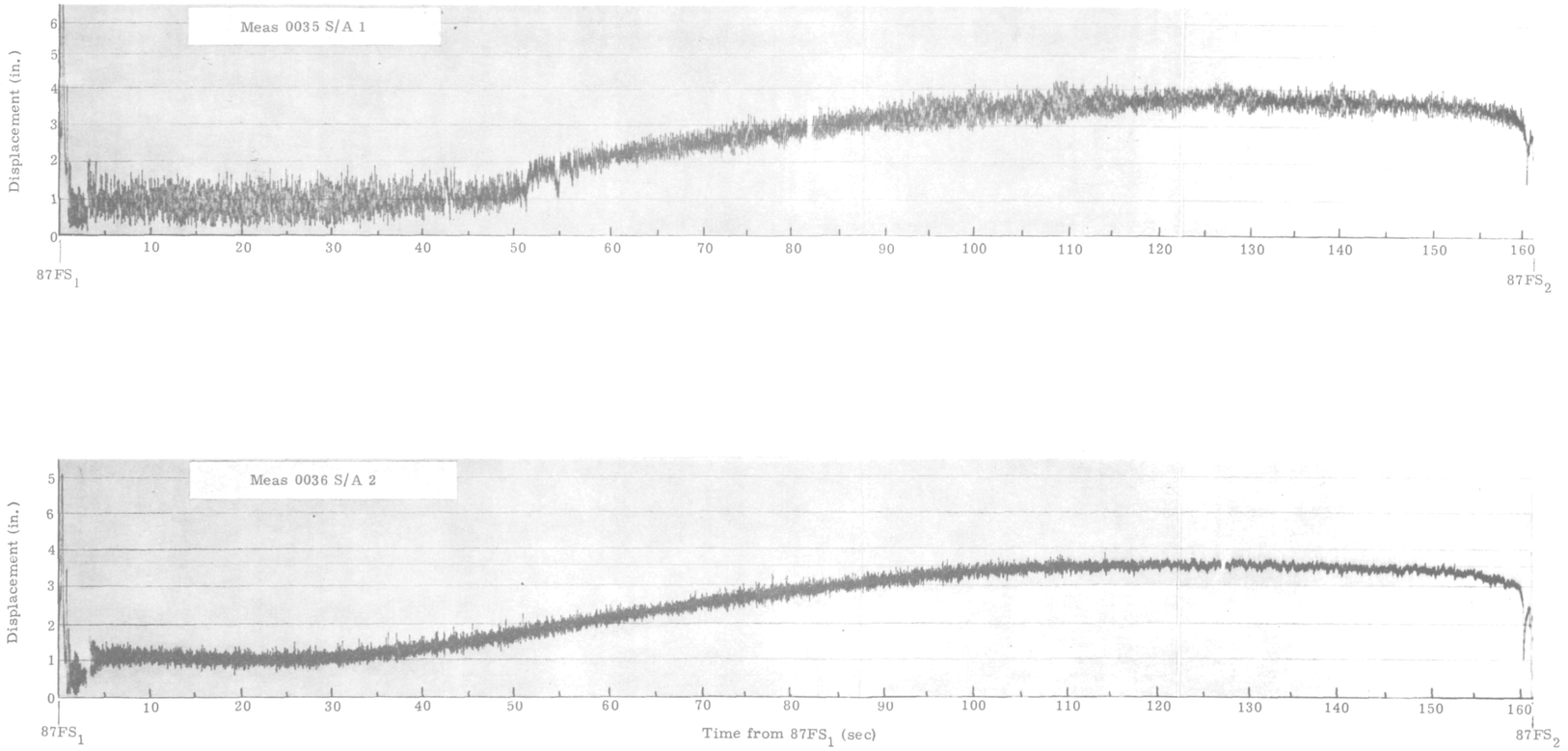

Fig. III-34. Fuel Accumulator Piston Travel

TABLE III-40

Dynamic Friction Levels for Dry Accumulators

S/A	Serial No.	Peak-toPeak Friction* (psi)	
		Bench	Preflight
1	B 014	0.7	0.8
2	B 017	1.1	1.0

*Maximum acceptable value = 2.0 psi

6. POGO Performance

The Stage I longitudinal oscillation level between LO + 100 seconds and BECO was very low. Flight data do not indicate significant structural responses in propulsion measurements at any time. In particular, oxidizer suction pressure (Meas 0017) and oxidizer standpipe pressure (Meas 0033 and 0034) do not increase in amplitude near BECO. Additional details on POGO appear in Chapter XII of this report.

D. PRESSURIZATION SUBSYSTEM

1. Prelaunch Pressurization

At approximately T-215 minutes, three of the four propellant tanks were pressurized, through AGE, from blanket pressure level to flight pressure levels. The Stage I fuel tank was not pressurized until T-200 minutes because the required hand valve was not open at T-215 minutes. The resultant time-pressure profiles (Fig. III-35) indicate that the process was normal. The tank ullage lockup pressures obtained from landline measurements made at T-0 and the related normal operating pressure ranges are presented in Table III-41.

TABLE III-41

Tank Ullage Lockup Pressures

Meas	Parameter	Normal Range (psia)	Measured (psia)
4125	Stage I fuel tank	27.5 to 31.5	28.7
4129	Stage I oxidizer tank	30.5 to 34.5	32.6
4602	Stage II fuel tank	49.5 to 54.5	53.1
4605	Stage II oxidizer tank	53.5 to 57.5	55.8

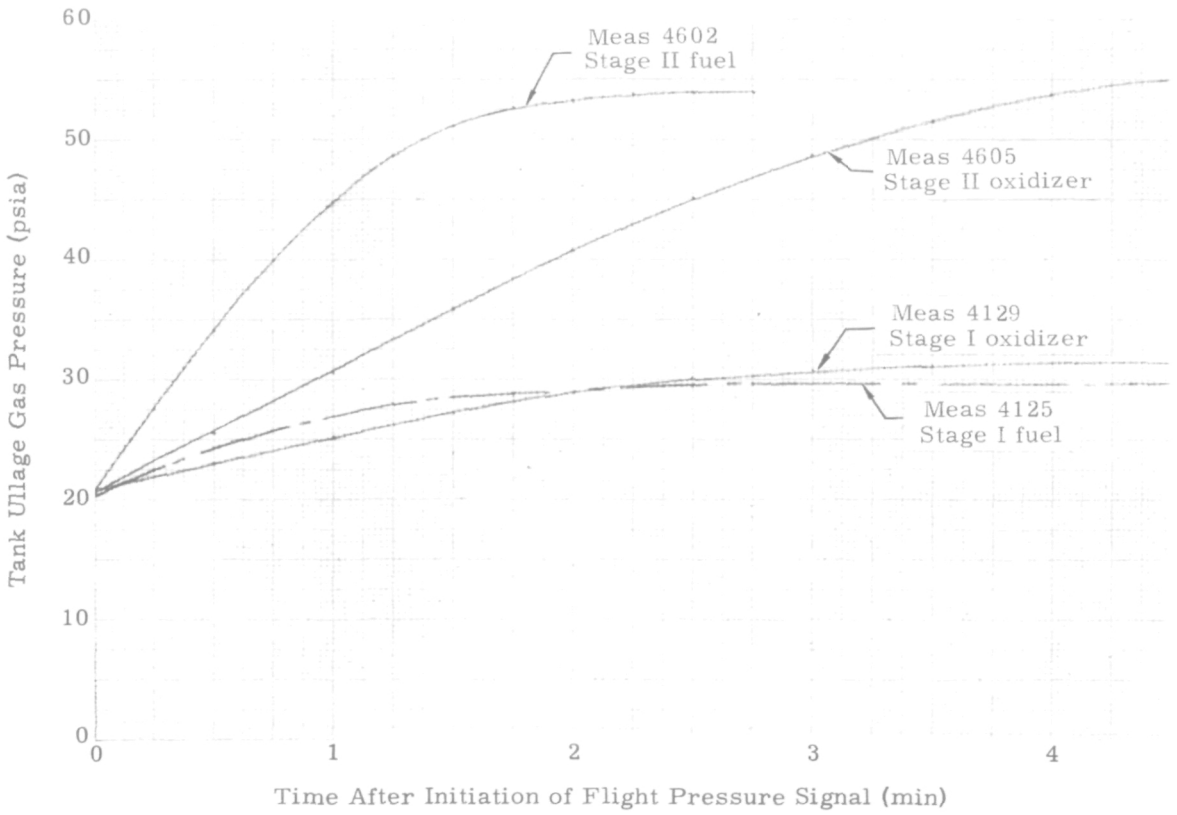

Fig. III-35. Tank Pressurization Cycle (blanket to flight pressure)

2. Flight Pressurization

Stages I and II ullage gas pressure time histories appear in Figs. III-36 through III-39. These plots show flight-measured pressures, preflight predicted pressures and postflight reconstructed pressures. The flight-measured pressures were obtained by averaging the telemetered output from each pair of pressure transducers in the individual tanks. The preflight predicted curves were taken from Ref. 10. The postflight reconstruction was based on flight measured values of engine performance, propellant temperatures and propellant loadings. A comparison of significant pressurization system parameters taken at FS_1 + 100 seconds is given in Table III-42.

TABLE III-42

Comparison of Significant Pressurization System Parameters at FS_1 + 100 Seconds

	Preflight Predicted	Flight Data	Postflight Reconstructed
Stage I Fuel Tank ①			
Tank pressure, P_{FT} (psia)	22.1	22.6	22.4
Nozzle inlet temperature, T_{FPOI} (°F)	245.0	--	212.0 ②
Flow ratio, W_{FP}/Q_{FS} (lb/ft^3)	0.06302	--	0.06428
Stage I Oxidizer Tank ③			
Tank pressure, P_{OT} (psia)	18.8	17.6	17.7
Orifice inlet specific enthalpy, H_{OPOI} (Btu/lb)	334.9	--	337.8 ②
Flow ratio, W_{OP}/Q_{OS} (lb/ft^3)	0.17422	--	0.17391
Stage II Fuel Tank ④			
Tank pressure, P_{FT} (psia)	50.6	48.7	48.9
Nozzle inlet temperature, T_{POI} (°F)	225.0	--	232.0 ②

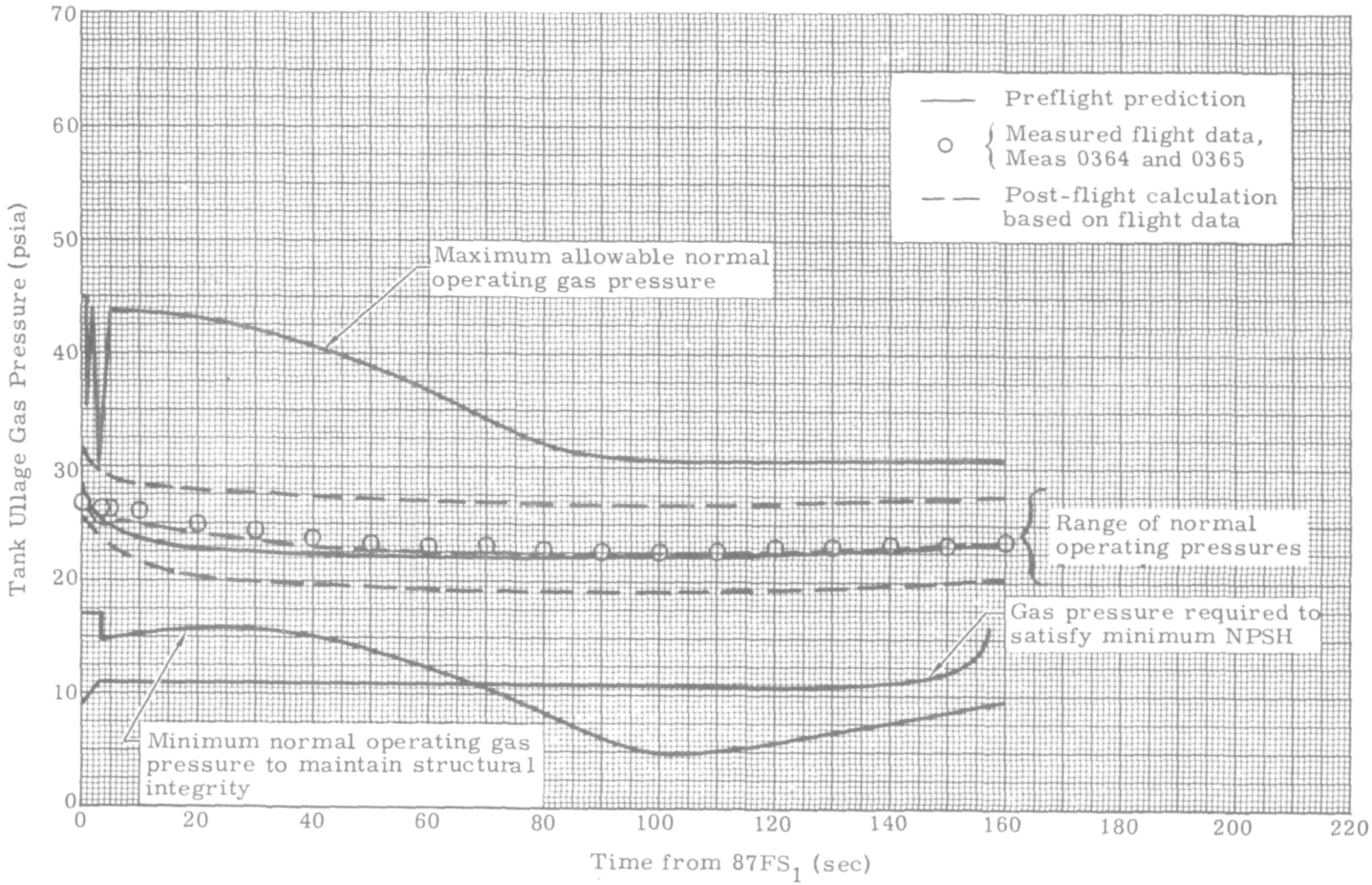

Fig. III-36. Stage I Fuel Tank Ullage Pressure History

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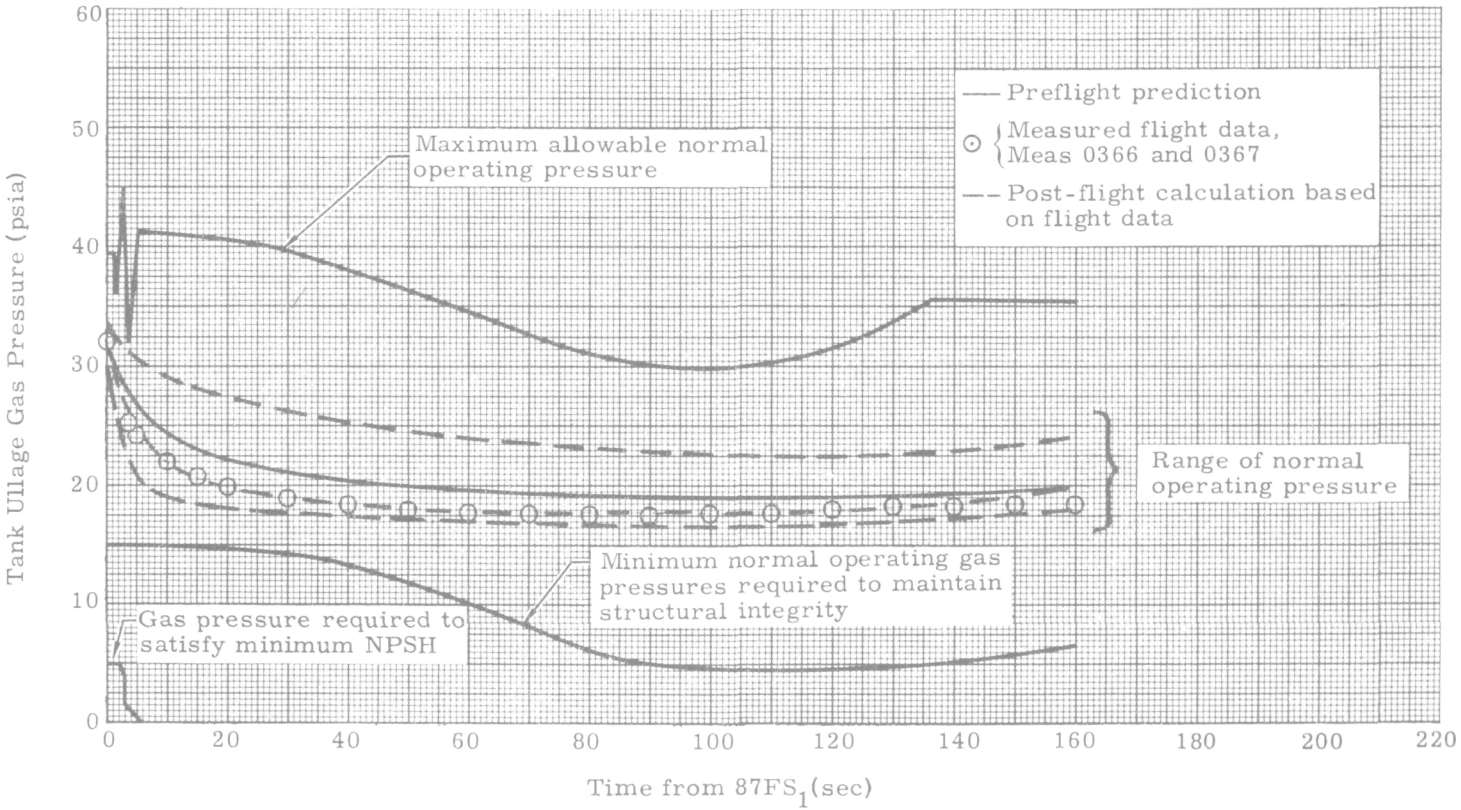

Fig. III-37. Stage I Oxidizer Tank Ullage Pressure History

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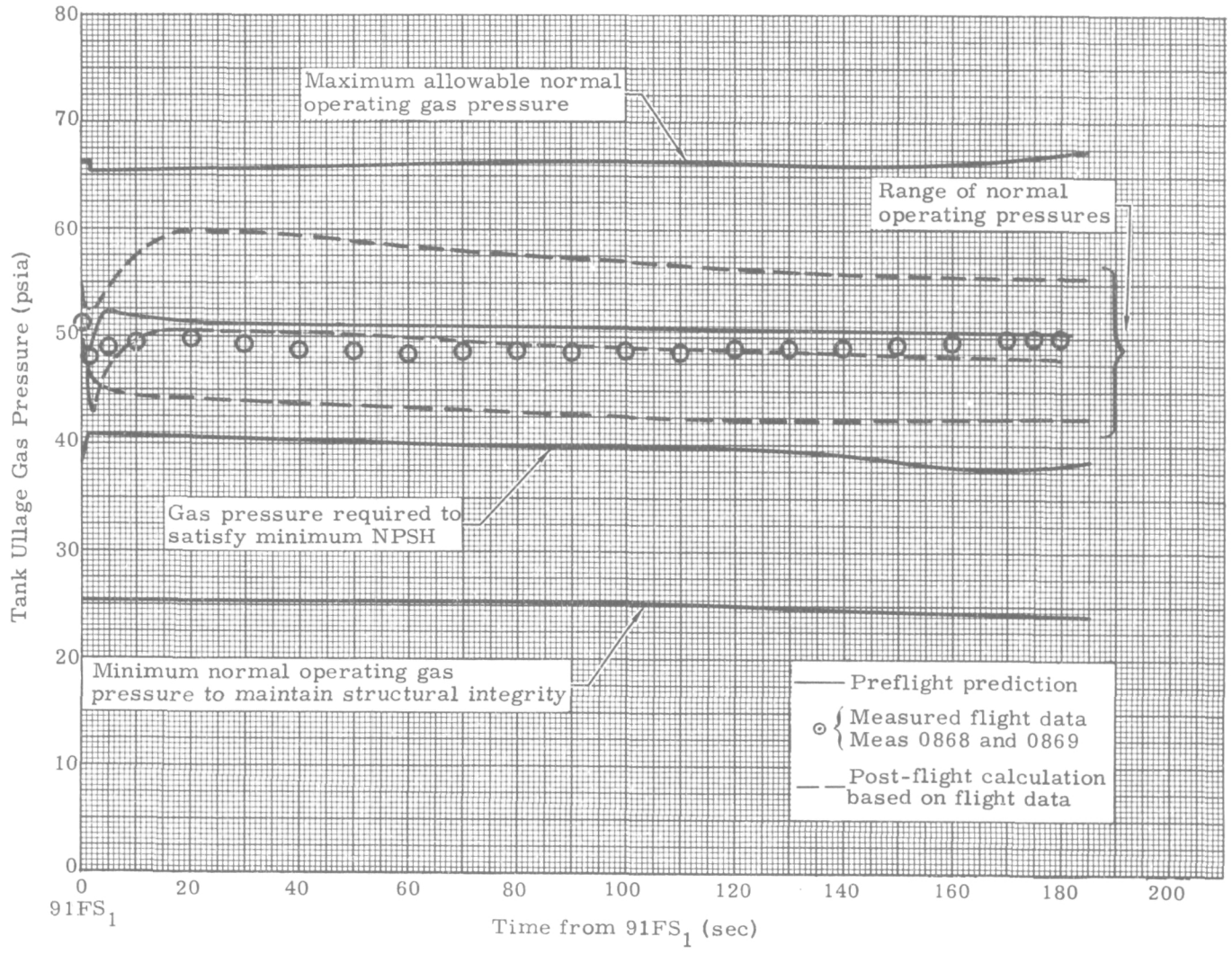

Fig. III-38. Stage II Fuel Tank Ullage Pressure History

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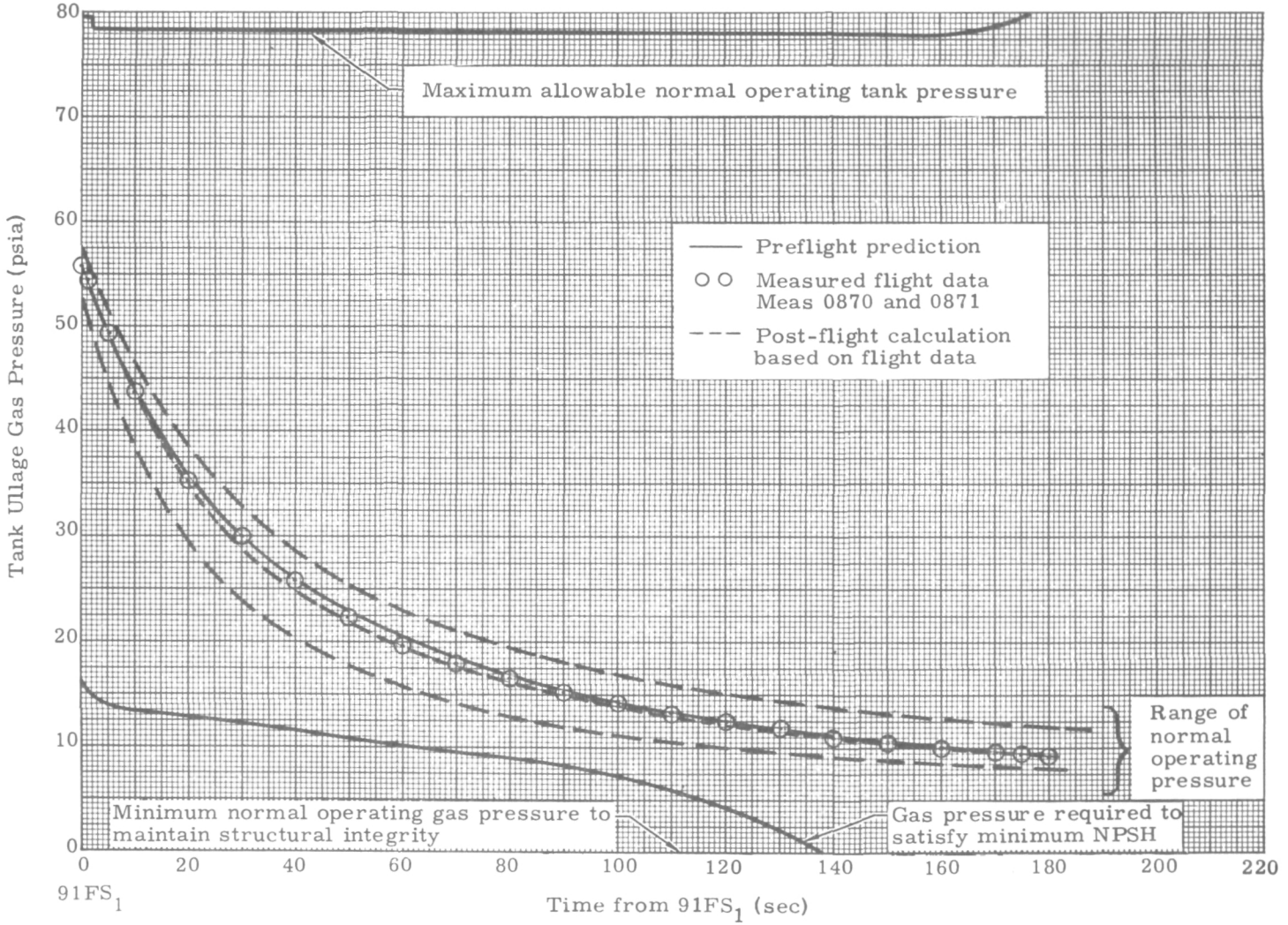

Fig. III-39. Stage II Oxidizer Tank Ullage Pressure History

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TABLE III-42 (continued)

	Preflight Predicted	Flight Data	Postflight Reconstructed
Flow ratio, W_{FP}/Q_{FS} (lb/ft^3)	0.14728	--	0.14425
Stage II Oxidizer Tank			
Tank pressure, P_{OT} (psia)	14.3	14.2	14.1
Propellant flow rate, Q_{OS} (ft^3/sec)	2.26317	--	2.25984

1 Nozzle diameter, FPN Stage I fuel--0.480 in.

2 Estimated, temperature not instrumented.

3 Flow control Venturi coefficient--0.0509.

4 Nozzle diameter, FPN Stage II fuel--0.260 in.

Figures III-40, III-41 and III-42 present the preflight-predicted and the in-flight-estimated pressurization parameters at the orifice or nozzle inlet.

3. Component Performance

All tank pressure sensors functioned normally. The maximum and mean pressure differences between pairs of sensors in each of the individual propellant tanks are shown in Table III-43.

TABLE III-43

Pressure Difference Between Tank Pressure Transducer Pairs

Tank	Maximum Difference (psi)	Mean Difference (psi)	Maximum Allowable Difference (psi)
Stage I oxidizer	0.28	0.13	1.50
Stage I fuel	0.34	0.11	1.50
Stage II oxidizer	1.04	0.40	2.25
Stage II fuel	0.48	0.33	2.25

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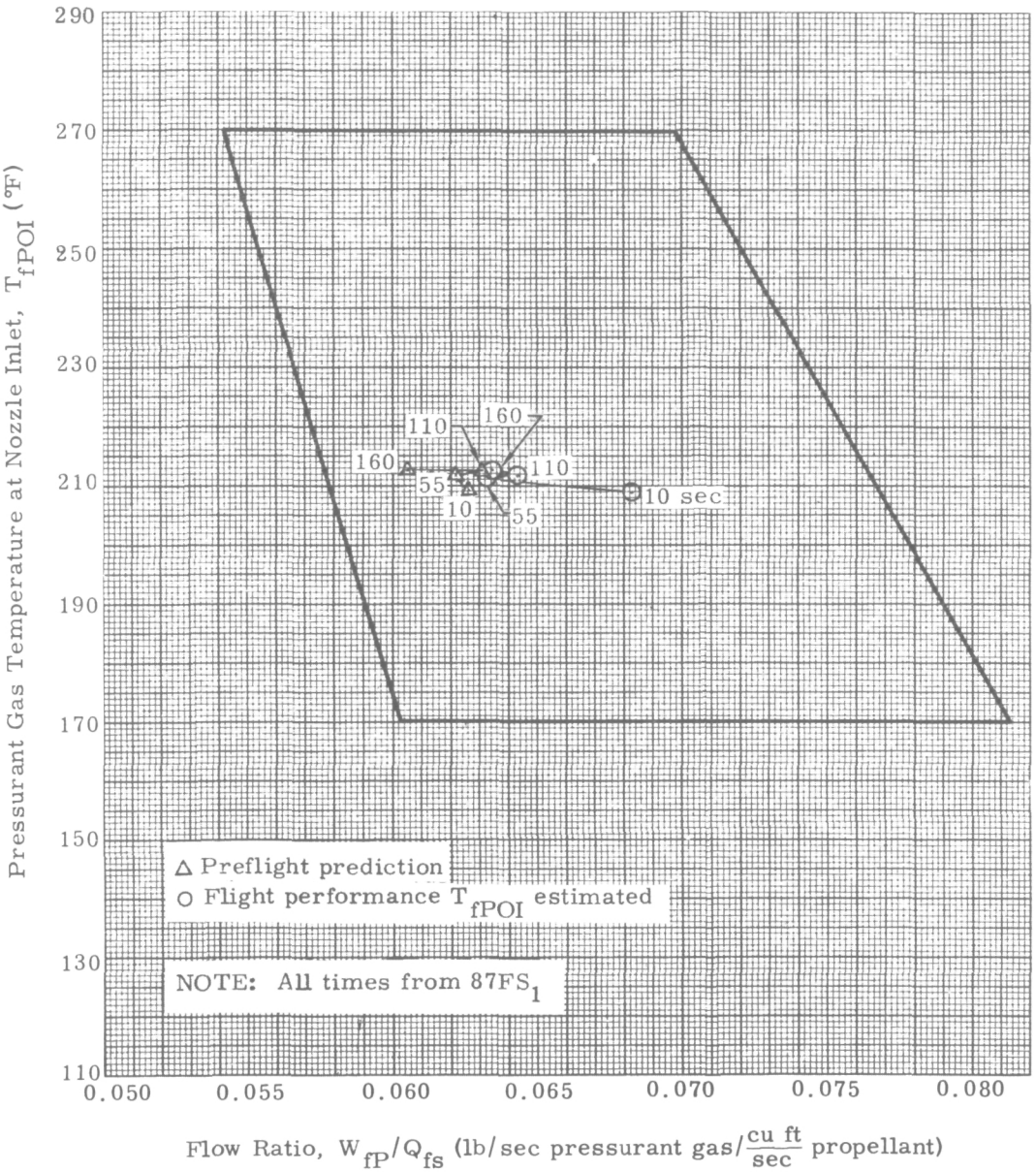

Fig. III-40. Stage I Fuel Tank Pressurant Performance

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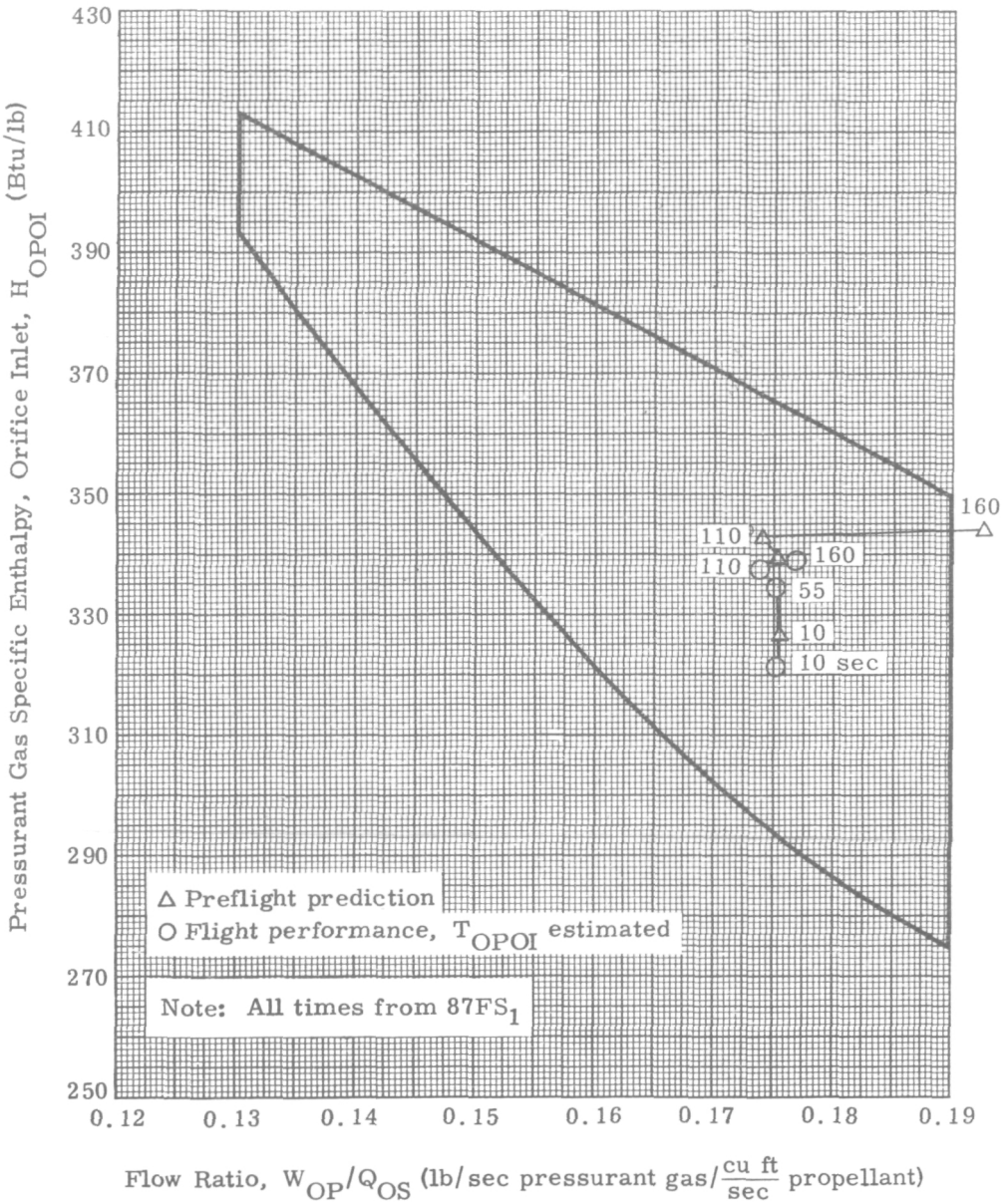


Fig. III-41. Stage I Oxidizer Tank Pressurant Performance

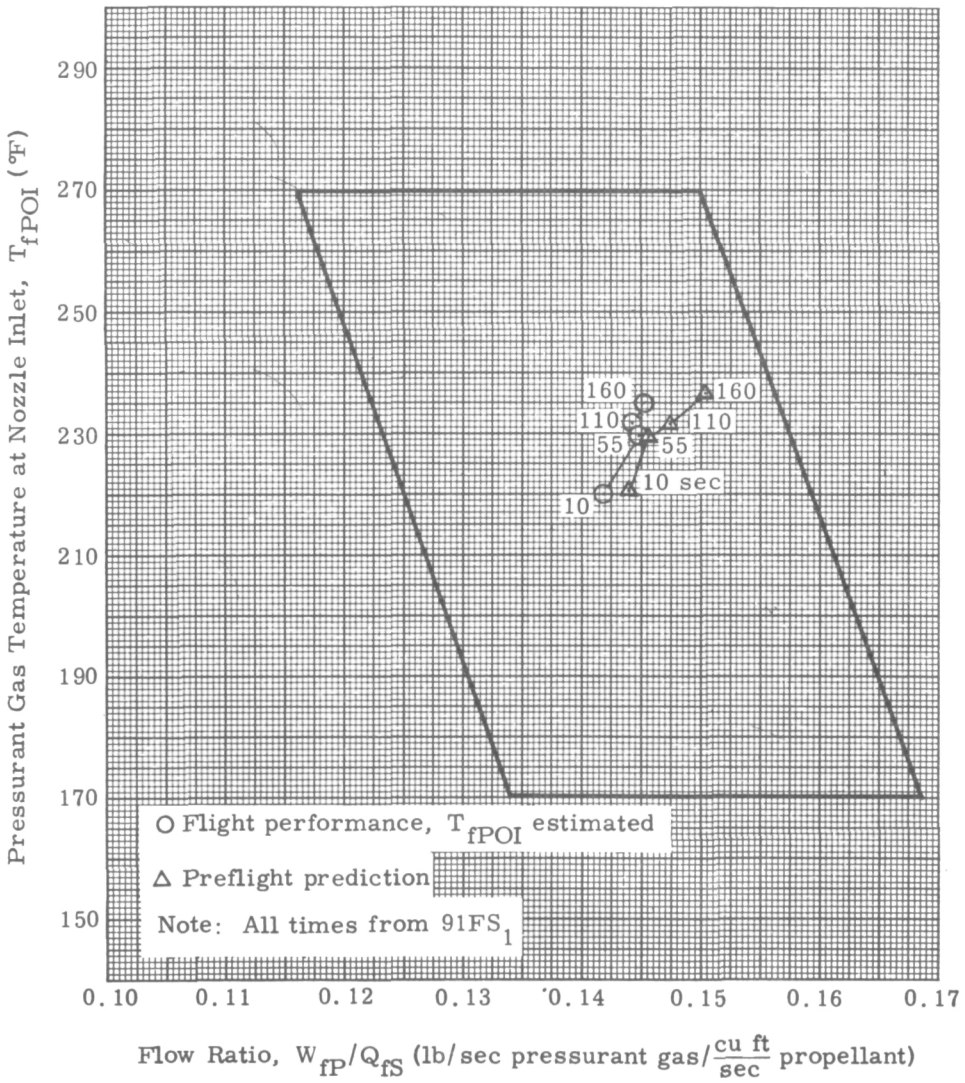


Fig. III-42. Stage II Fuel Tank Pressurant Performance

E. ENVIRONMENTAL CONTROL

1. Launch Vehicle Air-Conditioning System

This system, which serves launch vehicle Compartment 2 and all engine start cartridges, was operative continuously during the prelaunch activities until vehicle liftoff. The system operated satisfactorily. Table III-44 presents a summary of the system parameters.

TABLE III-44

Air-Conditioning System Performance Summary

Meas	Description	Observed Range	Specified Range	Remarks
4403	GLV supply air temperature	48° to 50.5° F	48° to 56° F (Compartment 2), 48° to 58° F (engine start cartridges)	Temperature of air supplied to GLV Compartment 2 and the engine start cartridges
4405	Compartment 2 supply air mass flow rate	Approximately 91 lb/min	82 lb/min (minimum)	
4418	Compartment 2 exhaust air temperature	55° to 61° F (61° F at liftoff)	40° to 75° F	Manual hold parameter
4045	Start cartridge temperature S/A 1	54.5° F (at liftoff)	35° to 74° F	S/N 0002510 manual hold parameter
4046	Start cartridge temperature S/A 2	56° F (at liftoff)	35° to 75° F	S/N 0002749 manual hold parameter
4612	Start cartridge temperature S/A 3	53° F (at liftoff)	35° to 83° F	S/N 0001455 manual hold parameter

IV. FLIGHT CONTROL SYSTEM

Analysis of the GT-6A flight control system (FCS) measured parameters indicated satisfactory system operation during both Stages I and II flights. The primary FCS was in command throughout and no switchover to the secondary system was required.

A. STAGE I FLIGHT

1. Ignition and Liftoff Transients

Peak actuator travels and rate gyro disturbances recorded during the ignition and holddown period are presented in Table IV-1. The 4_1 actuator transducer anomaly during this period inhibits accurate determination of 4_1 travel.

TABLE IV-1

Transients During Stage I Holddown Period

Actuator Designation	Maximum During Ignition		Maximum During Holddown Null Check (in.)
	Travel (in.)	Time from LO (sec)	
Pitch, 1_1	-0.122	-2.26	-0.03
Yaw/roll, 2_1	+0.209	-2.25	+0.01
Yaw/roll, 3_1	+0.169	-2.28	-0.01
Pitch, 4_1	--	--	--
Axis	Maximum Rate, Stage I Gyro (deg/sec)		Maximum Rate, Stage II Gyro (deg/sec)
Pitch	0.5		0.5
Yaw	0.7		0.6
Roll	1.1		--

The combination of thrust and engine misalignments at full thrust initiated a roll transient at liftoff. The FCS response to correct these misalignments, shown in Fig. IV-1, kept the roll rate to a maximum of 1.9 deg/sec clockwise (CW) at 0.14 second after liftoff. The rate oscillation, at a frequency of 4.5 rad/sec, damped out in 1.8 seconds. A roll bias of 0.69 degree CW was introduced at liftoff by the equivalent engine misalignment of 0.14 degree.

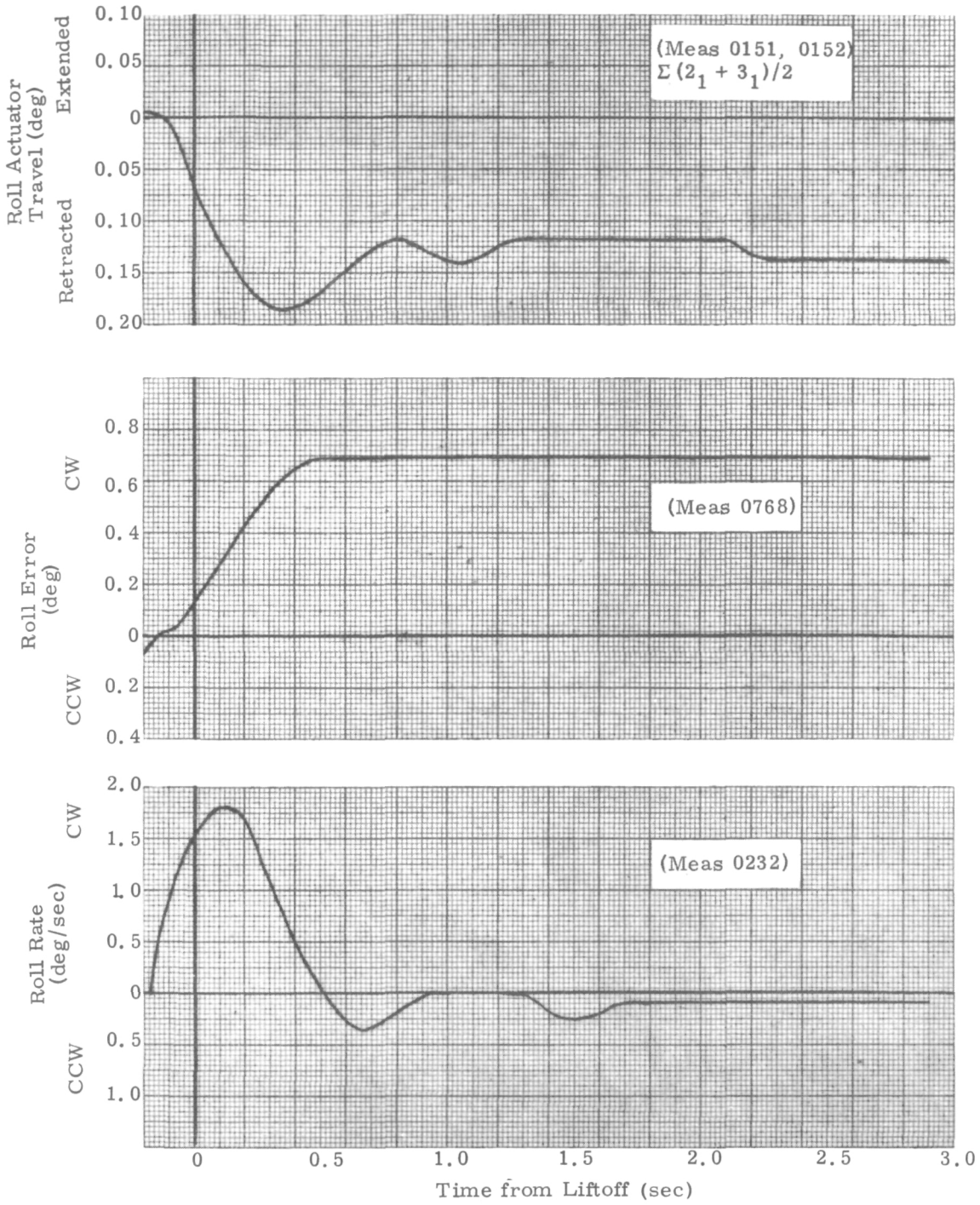


Fig. IV-1. Liftoff Roll Transients

The pitch and yaw liftoff transients indicated by the rate gyros were: Stage I gyro 0.5 deg/sec and Stage II gyro 0.3 deg/sec for pitch and Stage I gyro 0.6 deg/sec and Stage II gyro 0.2 deg/sec for yaw.

2. Roll and Pitch Programs

The TARS roll and pitch programs performed nominally as shown in Table IV-2. The maximum roll and pitch overshoots which occurred at the initiation of their respective programs were 1.6 deg/sec clockwise for roll and 0.9 deg/sec nose-down for pitch.

TABLE IV-2

TARS Roll and Pitch Programs

Program	Time from LO (sec)	Nominal Time (sec)	Rate Gyro Average (deg/sec)	Torquer Monitor Indication (deg/sec)	Nominal Rate (deg/sec)
Roll					
Start	17.66	17.68	+1.24	+1.25	+1.25
Stop	20.46	20.48			
Pitch Step 1					
Start	22.99	23.04	-0.70	-0.68	-0.709
Pitch Step 2					
Start	88.20	88.32	-0.52	-0.55	-0.516
Pitch Step 3					
Start	118.83	119.04	-0.20	-0.25	-0.235
Stop	161.70	162.56			

3. TARS-IGS Comparison (Stage I)

The TARS and IGS attitude error signals during Stage I flight for the pitch, yaw and roll axes are presented in Figs. IV-2, IV-3, and IV-4, respectively. The dispersion between the TARS and IGS signals was caused by a combination of TARS gyro and IGS-IMU drifts, errors in open-loop guidance programs, and reference axis cross-coupling. The dispersion (TARS attitude minus IGS attitude) at BECO was -0.49 degree in the pitch axis, +0.18 degree in the yaw axis and +1.26 degrees in the roll axis.

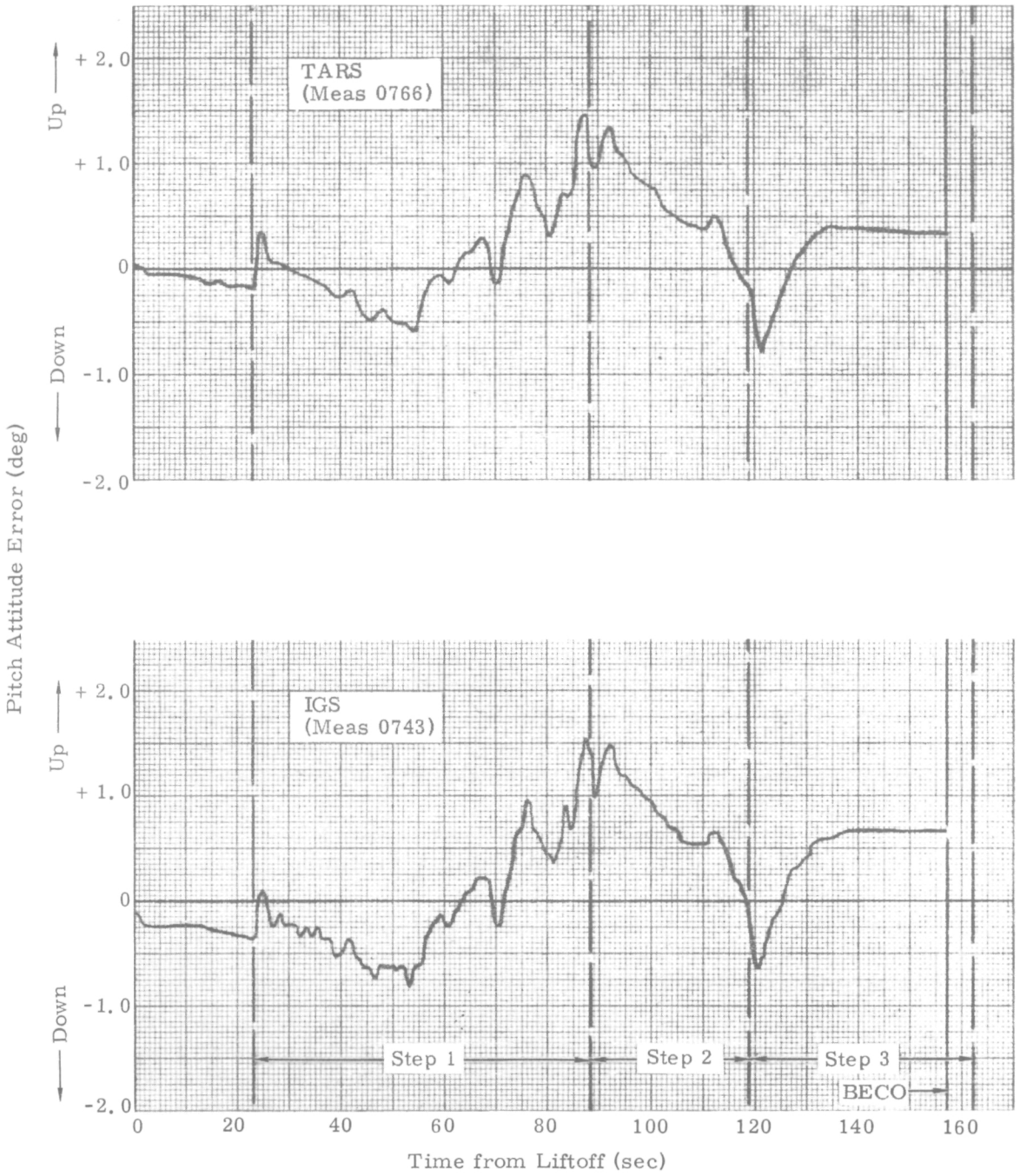


Fig. IV-2. Pitch Attitude Error History During Stage I Flight

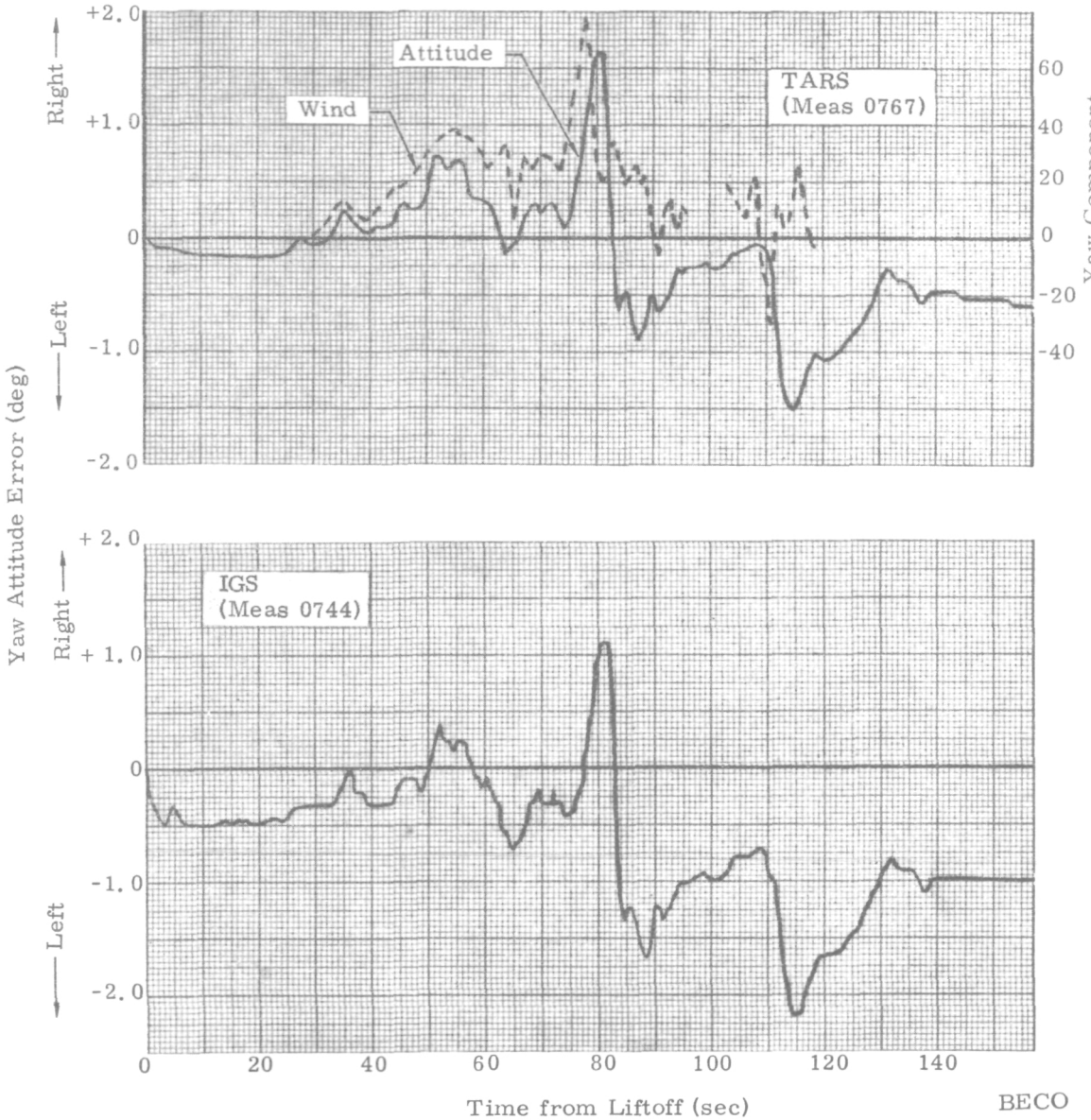


Fig. IV-3. Yaw Attitude Error History During Stage I Flight

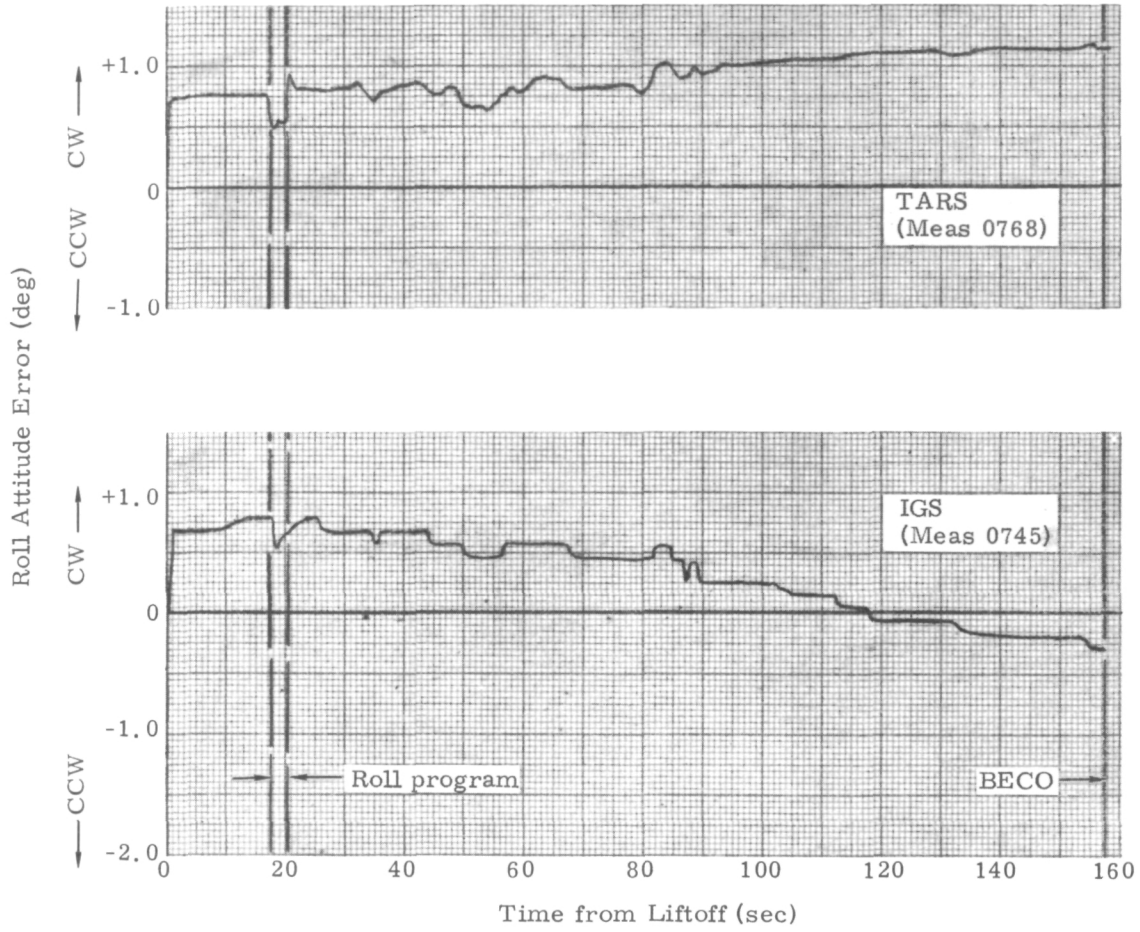


Fig. IV-4. Roll Attitude Error History During Stage I Flight

4. Stage I Flight Disturbances

Vehicle disturbances during flight were caused by prevailing winds aloft. The speed and direction of the winds are plotted in Fig. II-25.

The flight control system response to these disturbances was normal and well controlled. The TARS and IGS attitude error signals during Stage I flight for the pitch, yaw and roll axes are presented in Figs. IV-2, IV-3 and IV-4, respectively. In addition, a comparison of the yaw component of wind and the yaw attitude error is shown in Fig. IV-3. The maximum rates and attitude errors recorded during Stage I flight are shown in Table IV-3.

During the wind disturbances, oscillations between 1.0 and 1.6 rad/sec with an average peak-to-peak overshoot amplitude of less than 0.2 degree of attitude error occurred in pitch and yaw at the predicted GT-6 rigid body oscillatory mode frequencies, varying with flight condition. Since the level of pitch and yaw excitation was of large magnitude, there was evidence of inertial coupling producing excitation in the roll channel and thereby causing maximum peak-to-peak roll oscillations of 0.35 degree.

The time for control system gain change on GT-6A was changed from LO + 105 seconds to LO + 110 seconds to improve stability margin. At the actual time of gain change (LO + 109.8 seconds) there was a small, but highly damped, pitch transient which reached a maximum of 0.52 degree nose-up. Prior to gain change, the pitch attitude error was 0.35 degree nose-up. The attitude error in yaw at the time of gain change was less than 0.1 degree yaw left. The exact amount of yaw overshoot due to gain change cannot be determined since the vehicle was experiencing disturbances caused by changing winds at the time of gain change. The pitch transient overshoot was much less than for previous flights, on which the gain change occurred at LO + 105 seconds. There was no structural stability degradation because of the later gain change time on GT-6A.

5. Stage I Static Gains

As determined from telemetry data, the primary FCS static gains were within the instrumentation accuracy of preflight evaluations and indicate that no static gain deterioration was experienced during Stage I flight.

TABLE IV-3

Maximum Rate and Attitude Errors During Stage I Flight

Axis	Maximum Rates (deg/sec)*				Time from Liftoff (sec)			
	Primary Gyro		Secondary Gyro		Primary		Secondary	
	Stage I	Stage II	Stage I	Stage II	Stage I	Stage II	Stage I	Stage II
Pitch	+0.19 -1.12	+0.07 -1.01	+0.21 -1.00	+0.10 -1.11	2.0 88.0	0.2 87.9 & 68.3	0.6 87.8	0 → 10 88.0
Yaw	+0.39 -1.41	+0.49 -1.32	+0.40 -1.32	+0.50 -1.29	78.0 82.3	77.9 82.3	77.4 82.3	77.1 82.9
Roll	+1.88 -1.10		+1.90 -1.14		0.1 157.1		0.1 157.1	

Axis	Maximum TARS Attitude Error* (deg)	Time from Liftoff (sec)
Pitch	+1.48 -0.79	97.0 120.8
Yaw	+1.66 -1.53	80.5 114.8
Roll	+1.20 0.0	154.0 0.0

*Bias removed.

B. STAGE II FLIGHT

1. Staging Transients

During staging, moderate sustainer vehicle rates and attitude errors were observed. The maximum attitude error, measured from the pre-BECO level, and the maximum vehicle rates are given in Table IV-4.

2. Stage II Attitude Errors and Biases

The pitch and yaw attitude errors are shown in Figs. IV-5 and IV-6, respectively; the roll attitude error remained constant at -0.52 degree after staging transient. The predicted pitch and yaw attitudes account for the center-of-gravity displacement from the vehicle longitudinal axis and for the position of the roll thrust off the longitudinal axis. The additional biases from the predicted attitudes, -0.07 degree in pitch and +1.25 degrees in yaw, are caused by engine thrust vector misalignment due to structural deformation at the engine gimbal assembly. These biases are of the same magnitude noted on previous flights and are within predicted limits. The deviation from the biased predicted attitudes was due to system hysteresis and gain sensitivity.

3. Response to Radio Guidance Commands

The TARS timer generated the guidance enable command at LO + 161.64 seconds. Response to the first pitch command was at 168.33 seconds and consisted of a small down command followed by a 0.55 deg/sec down command for 1.05 seconds. After the first 24.4 seconds of pitch guidance, all ensuing pitch commands were less than 0.25 deg/sec. Response to the first yaw command was at 169.23 seconds and consisted of a small right command followed by a 0.25 deg/sec right command. After the first 31.5 seconds of yaw guidance, subsequent yaw commands were less than 0.06 deg/sec. The rate gyro signals substantiated correct response of the FCS to the guidance commands.

4. Stage II Static Gains

The primary FCS static gains as determined from telemetry data were within the instrumentation accuracy of preflight calculations.

C. POST-SECO FLIGHT

1. Vehicle Motions

Prior to SECO, the pitch actuator was retracted, producing a sustainer engine gimbal deflection of 0.19 degree to correct for a pitch error of -0.84 degree nose-down. In yaw, the error was +0.17 degree

TABLE IV-4

Maximum Staging Rates and Attitude Errors

Axis	BECO to Separation		Separation to Telemetry Blackout		Telemetry Blackout to Plus One Second	
	Maximum Rates (deg/sec)	Time from BECO (sec)	Maximum Rates (deg/sec)	Time from BECO (sec)	Maximum Rates (deg/sec)	Time from BECO (sec)
Pitch						
Primary	+0.87 -1.60	0.053 0.126	+2.36 -2.25	0.734 0.744	+0.19 -1.12	1.135 1.148
Secondary	+1.16 -2.04	0.053 0.145	+3.10 -2.45	0.734 0.744	+0.40 -1.31	1.148 1.133
Yaw						
Primary	+0.70 -0.71	0.078 0.068	+1.50 -1.52	0.742 0.730	+2.62	1.60
Secondary	+0.60 -0.48	0.078 0.068	+2.04 -1.69	0.750 0.740	+2.73	1.60
Roll						
Primary	+0.10 -2.19	0.179 0.154	-3.67	0.733	+0.70 -1.19	1.51 1.07
Secondary	+0.10 -2.20	0.179 0.154	-3.28	0.733	+0.59 -1.07	1.51 1.07

Axis	Attitude Error (deg)	Flight Time (sec)
Pitch	-0.21	158.5
Yaw	+2.76	159.7
Roll	-1.57	158.4

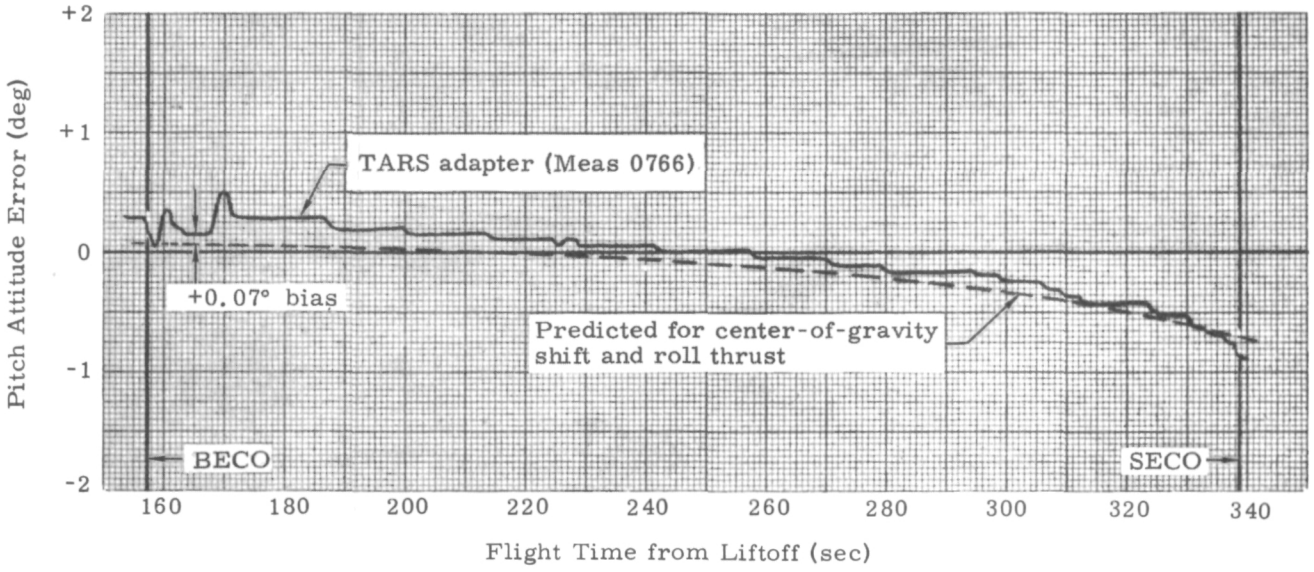


Fig. IV-5. Pitch Attitude Error During Stage II Flight

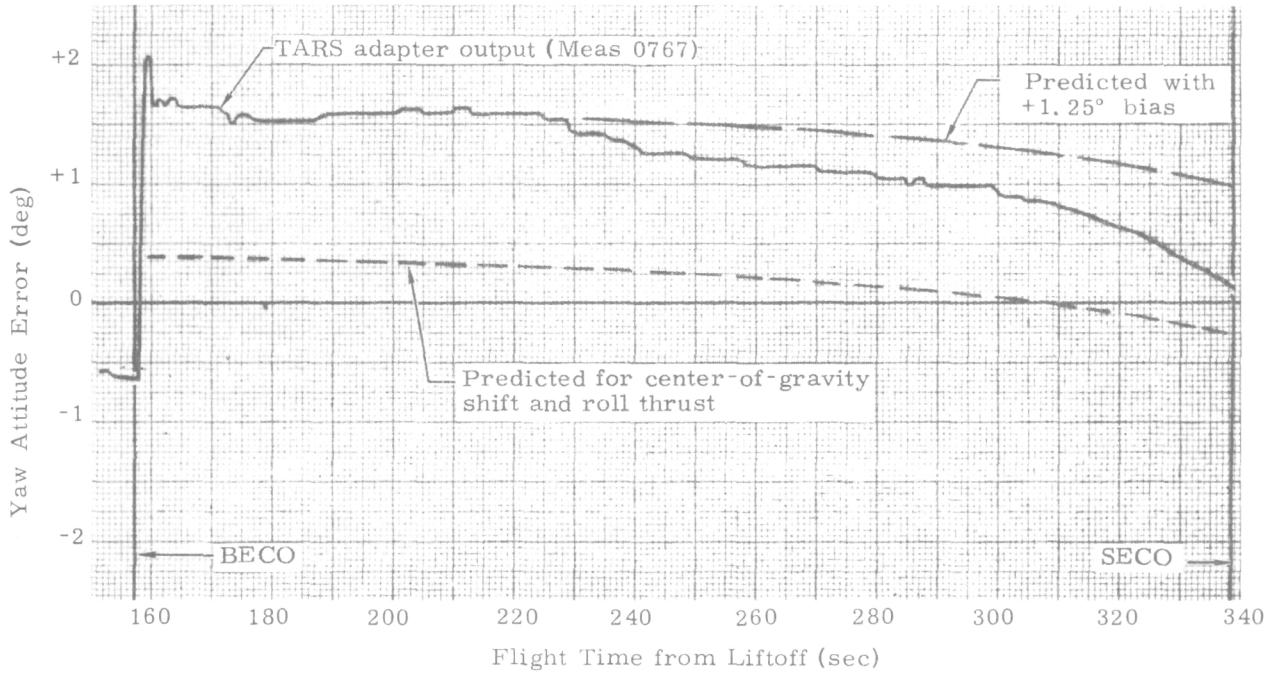


Fig. IV-6. Yaw Attitude Error During Stage II Flight

with an equivalent engine gimbal deflection of -0.07 degree. At SECO, the roll error was nearly zero.

Pitch, yaw and roll TARS and IGS attitude errors and rates, while operating on primary system during the period from SECO through spacecraft separation, are shown in Fig. IV-7. The maximum rates measured during the period following SECO appear in Table IV-5.

TABLE IV-5

Vehicle Rates Between SECO and Spacecraft Separation

Pitch Axis	Rate (deg/sec)
Maximum positive rate at $91FS_2$ + 2.4 sec	+0.59
Maximum negative rate at $91FS_2$ + 0.09 sec	-0.29
Rate at $91FS_2$ + 20 sec	-0.10
Rate at spacecraft separation ($91FS_2$ + 22.29 sec)	0.0
Yaw Axis	
Maximum positive rate at $91FS_2$ + 12.7 sec	+0.59
Maximum negative rate at $91FS_2$ + 3.7 sec	-1.11
Rate at $91FS_2$ + 20 sec	+0.49
Rate at spacecraft separation ($91FS_2$ + 22.29 sec)	+0.49
Roll Axis	
Maximum positive rate at $91FS_2$ + 2.3 sec	+0.49
Maximum negative rate at $91FS_2$ + 5.4 sec	-0.39
Rate at $91FS_2$ + 20 sec	+0.39
Rate at spacecraft separation ($91FS_2$ + 22.29 sec)	+0.19

Successful spacecraft separation was accomplished at 22.29 seconds after $91FS_2$. Vehicle rates did not exceed 1.1 deg/sec and the sustainer residual thrust at SECO + 20 seconds was less than 60 pounds. Successful spacecraft separation could have been accomplished at SECO + 20 seconds.

[redacted]

IV-14-2

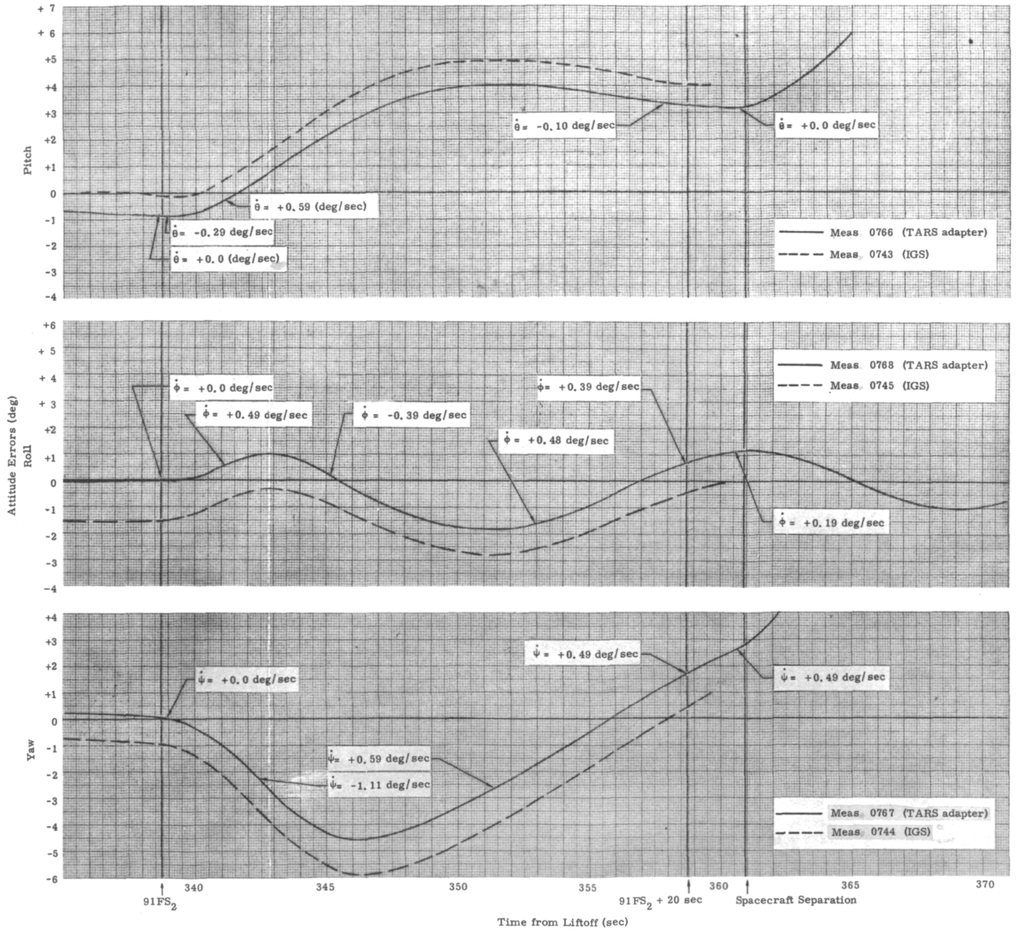


Fig. IV-7. Pitch, Roll and Yaw Attitude Errors During Post-SECO Flight

2. Post-SECO Transients

Low magnitude vehicle disturbances were measured on the low level axial accelerometer (Meas 0699) at $91FS_2$ + 5.1 seconds and again at $91FS_2$ + 6.6 seconds. During the first disturbance, the maximum change in axial acceleration was less than 0.01 g, peak-to-peak; the second disturbance caused a maximum change of less than 0.02 g peak-to-peak. These phenomena were not similar to the conventional post-SECO disturbances as exhibited on GT-1, -2, -4 and on various Titan II flights. The two perturbations had no discernible effect on any of the FCS parameters.

A third disturbance of larger magnitude occurred at $91FS_2$ + 17.49 seconds and indicated a maximum peak-to-peak acceleration of 0.089 g. The measured actuator and vehicle motions during the time of this disturbance are shown in Fig. IV-8. The axial acceleration response indicated an initial pulse followed by a few milliseconds of damping then a second pulse followed again by damping. This disturbance resembled the conventional post-SECO phenomenon, only in that the common double pulse was observed.

At the time of the disturbance, a definite change of level was noted on yaw actuator 5_2 with an equivalent -0.024-degree engine deflection. There were no discernible changes in vehicle rates, and it was impossible to determine the natural resonant response frequency from the axial accelerometer trace alone. On the basis of the 5_2 actuator motion, the engine gimbal hinge moment due to the disturbance was estimated to be 3283 in.-lb.

The fourth and largest disturbance was noted at $91FS_2$ + 27.95 seconds, or at spacecraft separation + 5.75 seconds. Time histories of this perturbation appear in Fig. IV-9. The maximum peak-to-peak acceleration was 0.093 g and the perturbation persisted for 0.28 second. In addition to the axial accelerometer transient, the disturbance was observed on the pitch (6_2) and yaw (5_2) actuators (0.05 and 0.097 degree, respectively, of equivalent engine deflection) as well as on the pitch and yaw gyros. The natural resonant response frequency measured was 7.7 cps. The peak-to-peak changes in pitch and yaw vehicle rates were 0.67 and 0.51 deg/sec, respectively. The equivalent engine gimbal hinge moment due to the disturbance was 14,892 in.-lb. Table IV-6 summarizes the post-SECO engine disturbances on all Gemini flights and various Titan II flights.

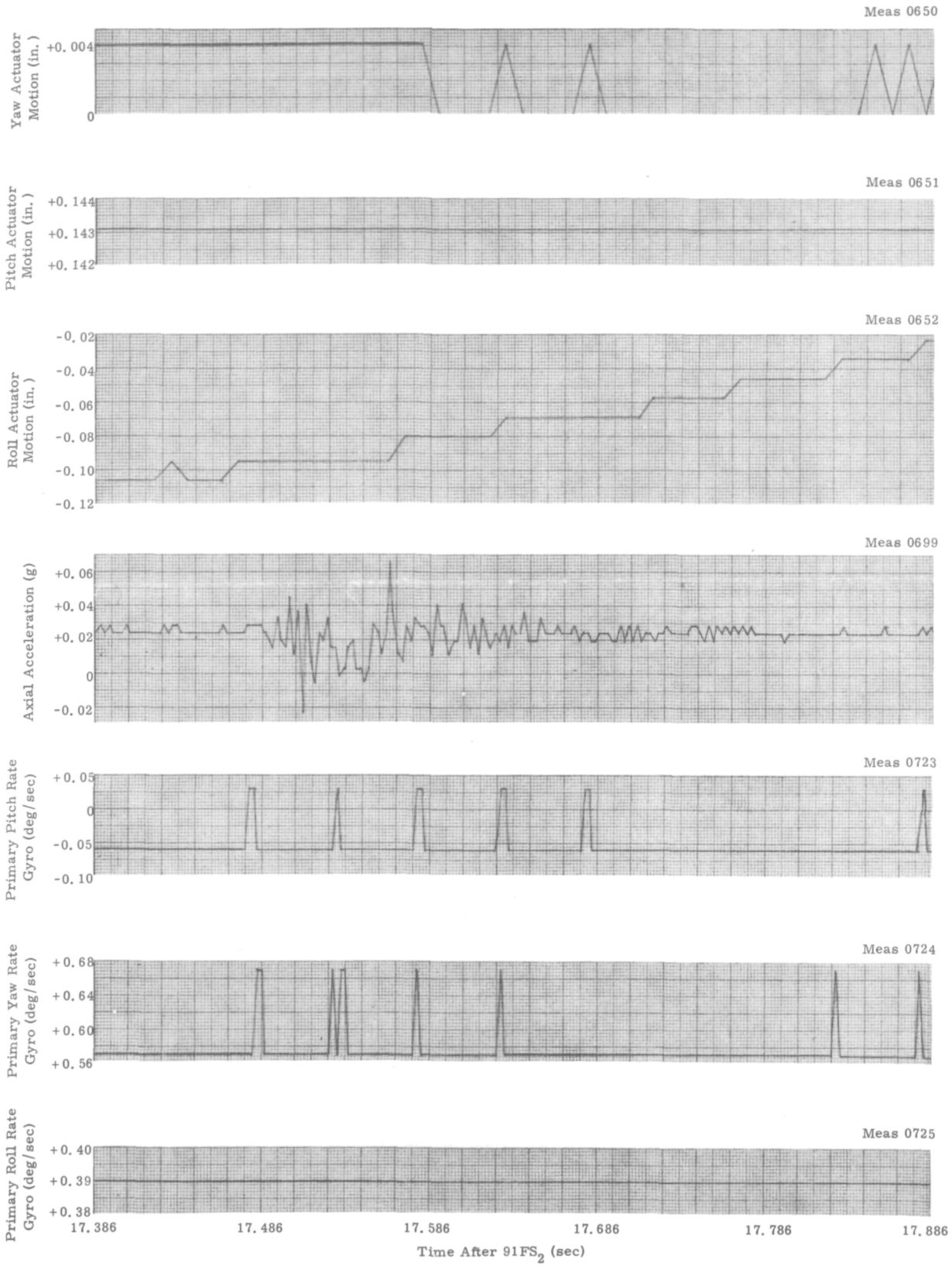


Fig. IV-8. Post-SECO Disturbance at $91FS_2$ + 17.49 Sec

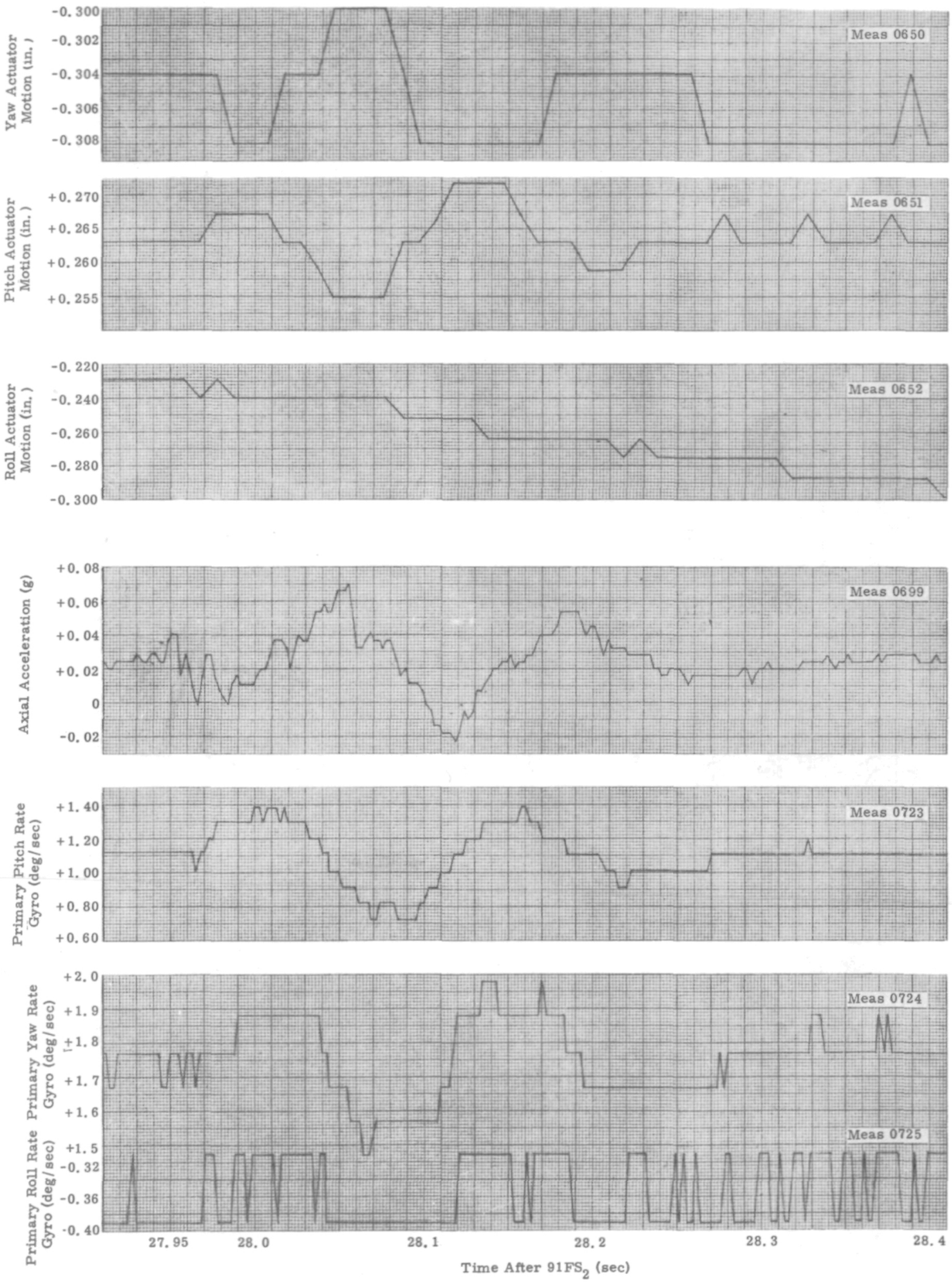


Fig. IV-9. Post-SECO Disturbance at $91FS_2$ + 27.95 Sec

TABLE IV-6

Post-SECO Engine Disturbances

Vehicle	Time From SECO (sec)	Actuator Travel Equivalent Gimbal (deg)		Differential Pressure (psid)		Estimated Engine Gimbal Hinge Moment (in. -lb)	Frequency (cps)
		δ_2	θ_2	δ_2	θ_2		
N5	5.25	+0.35	-0.40	N/A	N/A	57,400	5
N6	4.60	+0.20	-0.15	-850	-900	31,750	10
N8	5.44	+0.25	+0.20	-950	-1000	35,320	55
N12	8.1	+0.10	0	-500	-150	13,430	9
N16	3.1	-0.20	-0.20	+1300	+900	40,400	12.5
N17	8.53	-0.90	+0.35	+1600	-1000	48,400	5.6
N18	8.20	+0.10	+0.10	+100	-300		
N24	10.50	+0.10	--	-100	--	2,560	--
N25	5.00	+0.50	+0.30	-2000	-1700	50,000	5
N27	3.5	+0.15	-0.20	-2500	+2000	62,500	5
N31							
(1)	2.9	--	--	200	--	5,200	12.5
(2)	30.5	--	--	-150	--	3,900	8.3
B15	3.3	+0.35	--	N/A	N/A	--	--
N33	-20.	--	--	-3000	+3000	--	--
N3A	2.77	--	--	+1700	-1750	--	6.7
GT-1	11.5	+1.0p-p	0.70p-p	N/A	N/A	94,600	6.3
GT-2	6.5	+0.8	0.30p-p	N/A	N/A	61,300	12.5
GT-3	None observed						
GT-4	3.14	0.32p-p	0.16p-p	N/A	N/A	49,600	7.8
	10.85	0.16p-p	0.23p-p	N/A	N/A	38,800	6.3
GT-5	None observed						
GT-6A	17.49	-0.024	0	N/A	N/A	3,283	No distinct frequency noted
	27.95	0.05p-p	0.097p-p	N/A	N/A	14,892	7.7
GT-7	None observed						

D. PRELAUNCH ACTUATOR POSITION ANOMALY

At T-20.7 seconds, the Stage I pitch actuator (4_1) position indication suddenly changed from a normal 2.57 volts to 3.0 volts. The voltage gradually increased to 3.4 volts prior to engine ignition at which time the voltage increased to 3.74 volts and then suddenly decreased to 2.07 volts. This was followed by an increase to 2.4 volts at umbilical drop. From T-20.7 seconds to umbilical drop, the trace exhibited approximately 20 millivolts of noise. From umbilical drop to approximately LO + 16 seconds, the voltage output was 2.40 volts. The VDA command did not change until liftoff + 1.4 seconds. Comparing the original 2.57-volt transducer output to 2.40 volts after liftoff shows an apparent negative voltage shift of 0.17 volt, which is equivalent to 0.085 inch of actuator movement (the amount of actuator null shift caused by engine ignition is unknown).

This anomaly is still under investigation. The transducer-shutdown printed circuit board, from the CP 2644 chassis used during the countdown, has been X-rayed and functionally tested in the ASFTS. To date, no abnormal function has been found. Simulated failures of this circuit board have not completely duplicated the problem. For example, shorting the Q_{11} transistor collector-to-base will simulate the failure only up to the point of engine ignition.

A functional position transducer has been partially disassembled and breadboarded such that various junctions within the electrical circuitry are available as test points. Various failure mode simulations have shown that a variable resistance shunt between the primary and tertiary transformer coils could conceivably reproduce the anomaly. However, the transformer coils in question are not wound adjacently. The transformer of another unit is presently being disassembled to permit observation of all construction details.

V. HYDRAULIC SYSTEM

Analysis of the telemetered data revealed that GT-6A hydraulic systems performed satisfactorily during Stage I and Stage II flight.

Prior to SFT, the engine-driven hydraulic pumps were replaced with newly cleaned units, thereby minimizing the probability of contamination during vehicle systems tests requiring hydraulic power. The newly installed pumps were checked with a Gaussmeter to verify free motion of the compensator.

A. STAGE I

1. Primary Subsystem

The final Stage I hydraulic system pressure and level check in the countdown was performed automatically by the sequencer. At T-180 seconds, function control A-7 initiated the motor-driven pump run, which pressurized the secondary system. Approximately 70 seconds later, AGE, using the motor pump, automatically selected and pressurized the primary system. Electric motor pump pressure was a normal 3210 psia at T-0. Engine start transients, starting at $87FS_1$ + 0.76 second, produced flow demands which dropped primary pressure to 2530 psia at $87FS_1$ + 0.86 second. Pressure recovery occurred immediately, indicating proper pump compensator response. The pressure overshoot on recovery peaked at 3270 psia at $87FS_1$ + 1.14 seconds. A steady-state pressure of 3020 psia was reached at $87FS_1$ + 1.7 seconds. There were no significant pressure perturbations either at liftoff or during flight. Pressure decayed normally during flight to 2800 psia at staging.

Prior to T-110 seconds, the static reservoir level was 59.6% full, and it decreased to a normal 34.8% full at T-0. The level increased during flight to 50.3% full at staging. This 15.5% increase is a result of normal fluid expansion with increasing fluid temperature.

The steady-state reservoir levels and the level changes during system pressurization were normal.

Primary and secondary system pressures and pressure switch actuation points are shown in Fig. V-1. A comparison of primary system pressures for GT-7 and GT-6A launches during engine start and holddown is presented in Fig. V-2.

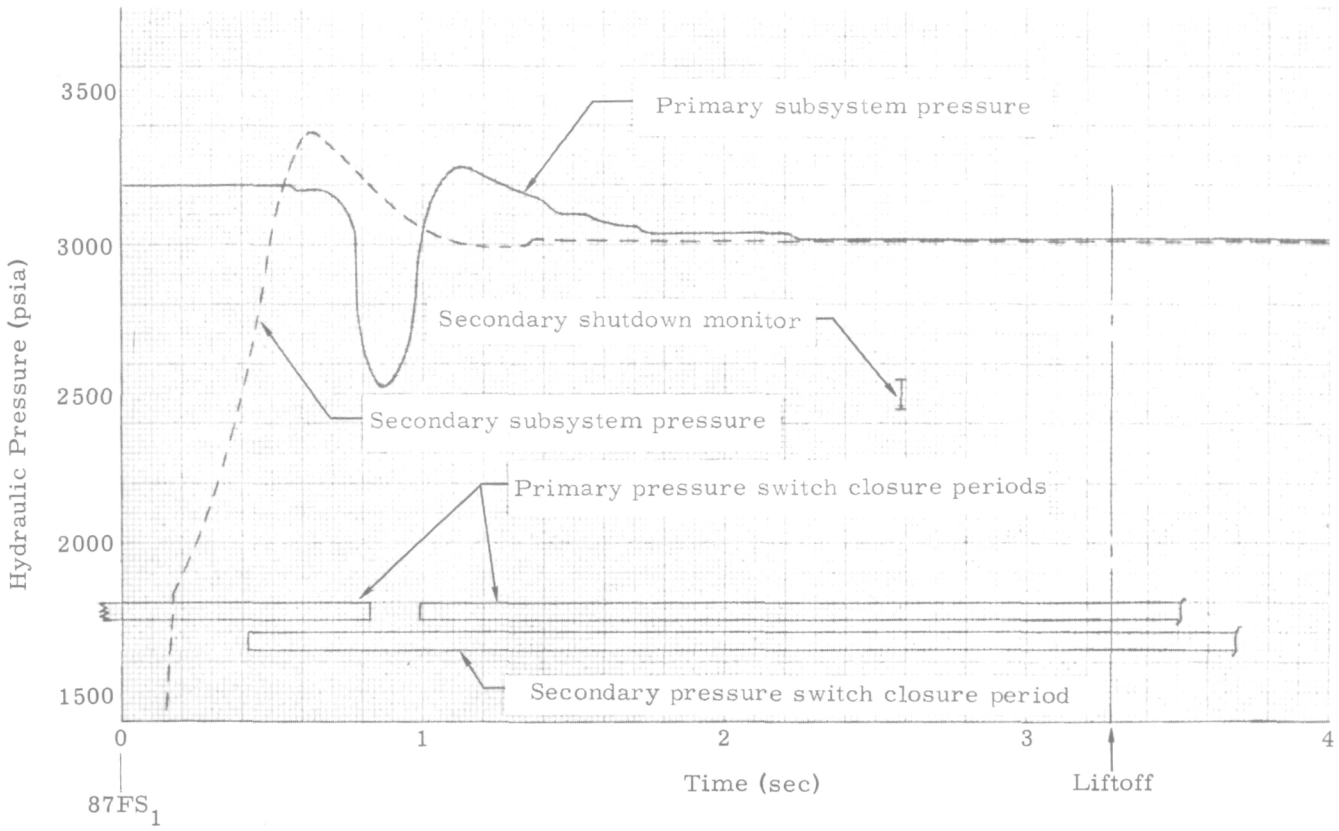


Fig. V-1. Stage I Hydraulic System Pressure Variation During Engine Start and Holddown

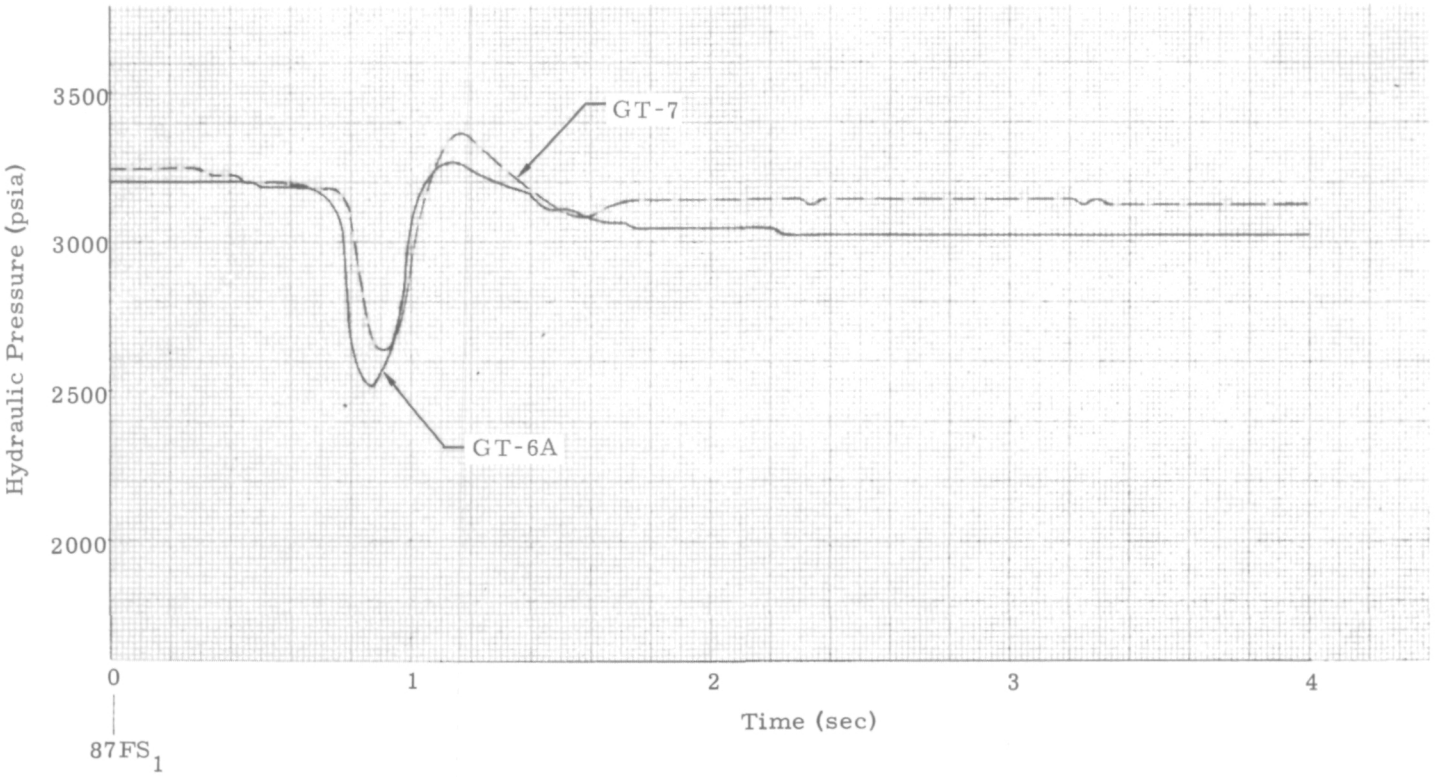


Fig. V-2. Comparison of Primary System Hydraulic Pressure During Engine Start and Holddown

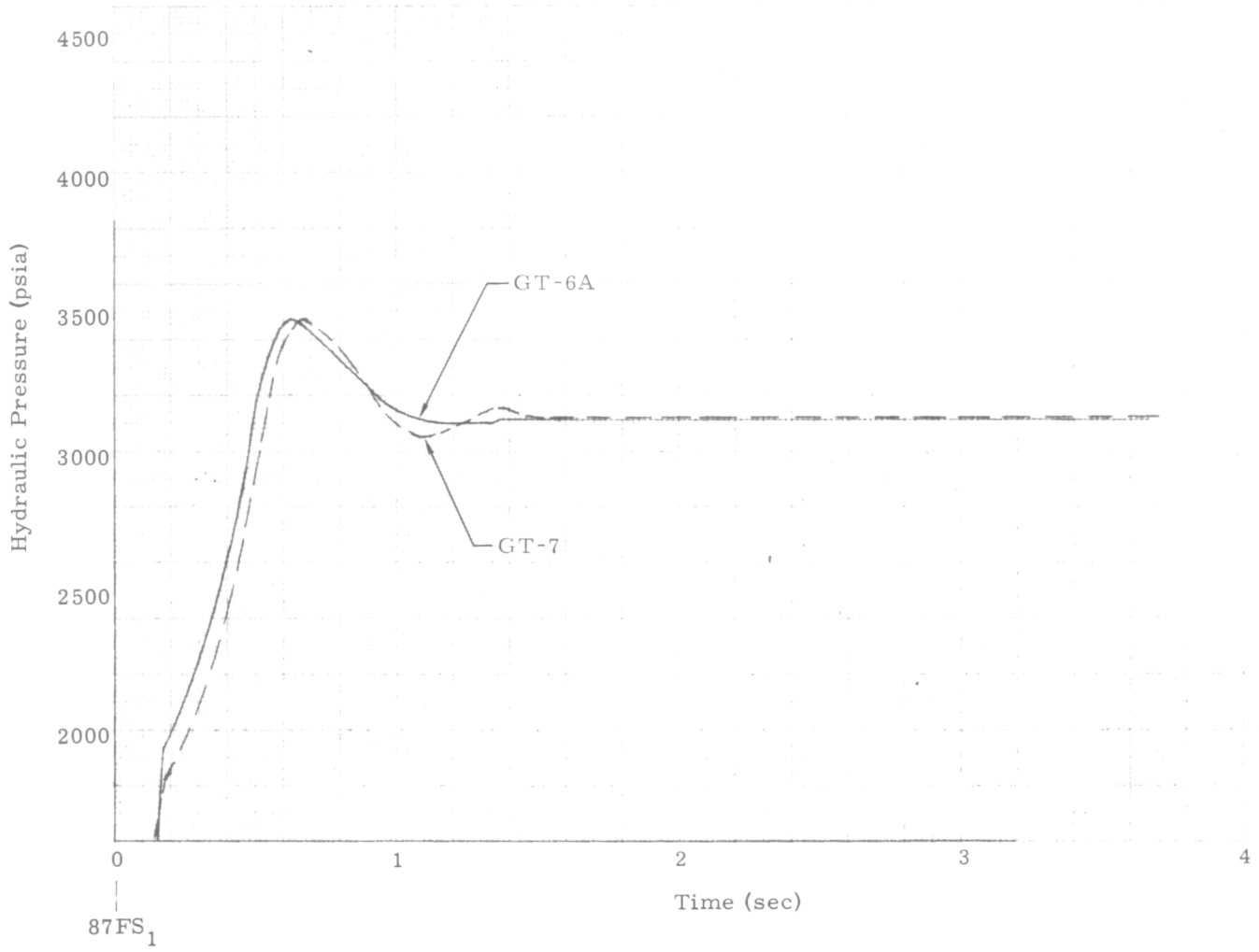


Fig. V-3. Comparison of Secondary System Hydraulic Pressure During Engine Start and Holddown

2. Secondary System

The final Stage I secondary hydraulic system pressure and reservoir level check was performed during a sequencer-initiated, motor-driven pump run from T-180 seconds to T-110 seconds. The indicated accumulator precharge was 1840 psia. Motor pump pressure was normal at 3200 psia at T-110 seconds.

The static reservoir level, which was a normal 55% full prior to pressurization at T-3 minutes, decreased to 28% full at T-110 seconds. These levels and the level changes during pressurization and depressurization of the system were normal.

At T-0, the system was unpressurized (soft). Pressure began to develop immediately as start cartridge energy rotated the engine turbine. Pressure overshoot reached a maximum of 3380 psia, indicating very good pump compensator response. A steady-state pressure of 3015 psia was reached at $87FS_1$ + 1.36 seconds. At the pressure shutdown interrogation point, the pressure remained steady at 3015 psia.

There were no pressure perturbations during flight since the system remained in a standby condition. Pressure decayed normally during flight to 2850 psia at staging.

The reservoir level stabilized at 34% full after engine start, increasing during flight to 42% full at staging. This 8% increase is a result of normal fluid expansion with increasing fluid temperature.

A comparison of secondary system pressures during engine start and holddown for GT-7 and GT-6A launches is presented in Fig. V-3.

B. STAGE II

The final Stage II hydraulic system pressure and level check was performed during a sequencer-initiated motor-driven pump run from T-240 seconds to T-180 seconds. The indicated accumulator precharge was 1800 psia. Electric motor pump pressure stabilized at a normal 3230 psia. The static reservoir level was 62.4% full, decreased to 36.2% full after pressure application and again increased to 61.0% full upon removal of pressure at T-3 minutes.

During engine startup at staging, the indicated accumulator precharge was 1800 psia, and pressure overshoot was to 3819 psia. Steady-state pressure after engine start was 2895 psia, decreasing to 2769 psia at SECO. No significant pressure perturbations occurred during flight. After SECO the pressure fluctuated with the engine rpm, a normal reaction to the low and variable turbine speeds occurring during this period.

The reservoir level was a normal 61.1% full prior to staging. After staging the level stabilized at 38.3% full, gradually increasing to 40.6% full at SECO. This 2.3% increase is normal.

The reservoir levels and changes during pressurization and depressurization of the system were normal.

VI. GUIDANCE SYSTEMS

A. RADIO GUIDANCE SYSTEM PERFORMANCE

1. Rate Beacon

Rate beacon performance was satisfactory. Good lock was maintained up to engine ignition and from approximately LO + 45 seconds to SECO + 40 seconds, except for the normal momentary loss of lock at Stage II engine ignition. The loss of lock at Stage I engine ignition is also considered normal; relock occurred as the primary antenna was brought into favorable ground station view.

Values of the rate beacon telemetered functions during flight are listed in Table VI-1.

TABLE VI-1

RGS Telemetered Functions

Function	Meas	Maximum Value	Minimum Value
Rate beacon			
Received signal No. 1	0750	4.18 vdc*	4.10 vdc*
Phase detector	0751	3.24 vdc*	2.80 vdc*
Power out	0752	4.18 vdc*	4.14 vdc*
30-volt supply	0746	2.84 vdc	2.80 vdc
Pulse beacon			
Magnetron current	0753	3.72 vdc**	3.66 vdc**
AGC	0754	-9.2 dbm**	-41.1 dbm**
15-volt supply	0747	4.18 vdc	4.14 vdc
Decoder			
10-volt supply	0748	4.48 vdc	4.44 vdc

*Does not include normal periods of unlock discussed in Section VI-A-1.

**Does not include antenna crossover period.

2. Pulse Beacon

Pulse beacon performance also was satisfactory. Good lock* was maintained through Stage I engine ignition and up to approximately SECO + 40 seconds.

Normal oscillations during the antenna crossover period were observed in AGC from approximately LO + 40 seconds to LO + 80 seconds. During this time, the minimum signal level received by the beacon was -51 dbm. A small percentage of messages were not received by the pulse beacon in the period from LO + 58.5 seconds to LO + 59.5 seconds. These misses occurred when the AGC oscillations were at peak (during the antenna crossover period). This condition has occurred on four of the seven Gemini flights to date and is considered normal. The normal ground station signal level increase occurred at LO + 90 seconds and was observed on telemetry to be approximately 20.8 dbm.

Values of the pulse beacon telemetered functions during flight are listed in Table VI-1.

3. Decoder

Decoder performance was satisfactory. Comparisons of the decoder telemetry data with the Burroughs computer-generated output data (10 pps) indicate that pitch and yaw steering signals and the SECO discrete were received and executed properly.

Values of the decoder telemetered functions are listed in Table VI-1.

4. Guidance Commands

a. Pitch steering

A profile of early closed-loop pitch steering in terms of Burroughs computer pitch steering commands, airborne decoder pitch steering commands, TARS gyro torquer monitor, and primary Stage II rate gyro is shown in Fig. VI 1. The decoder pitch steering output is also shown in Fig. VI-2 for the entire Stage II flight period.

TARS discrete No. 3 (RGS enable) was issued at approximately LO + 161.64 seconds, thereby energizing the airborne guidance initiate relay. At the same approximate time, pitch program No. 3 was terminated. This effect can be observed on Curves (c) and (d) of Fig. VI-1.

An initial decoder pitch-down command of about 0.10 deg/sec, lasting approximately 0.5 second, was issued at LO + 168.23 seconds. Following

*Good lock is defined as the condition in which no messages are missed by the pulse beacon. The ground station does not lose lock, however, unless a number of consecutive messages are missed.

VI-3-1

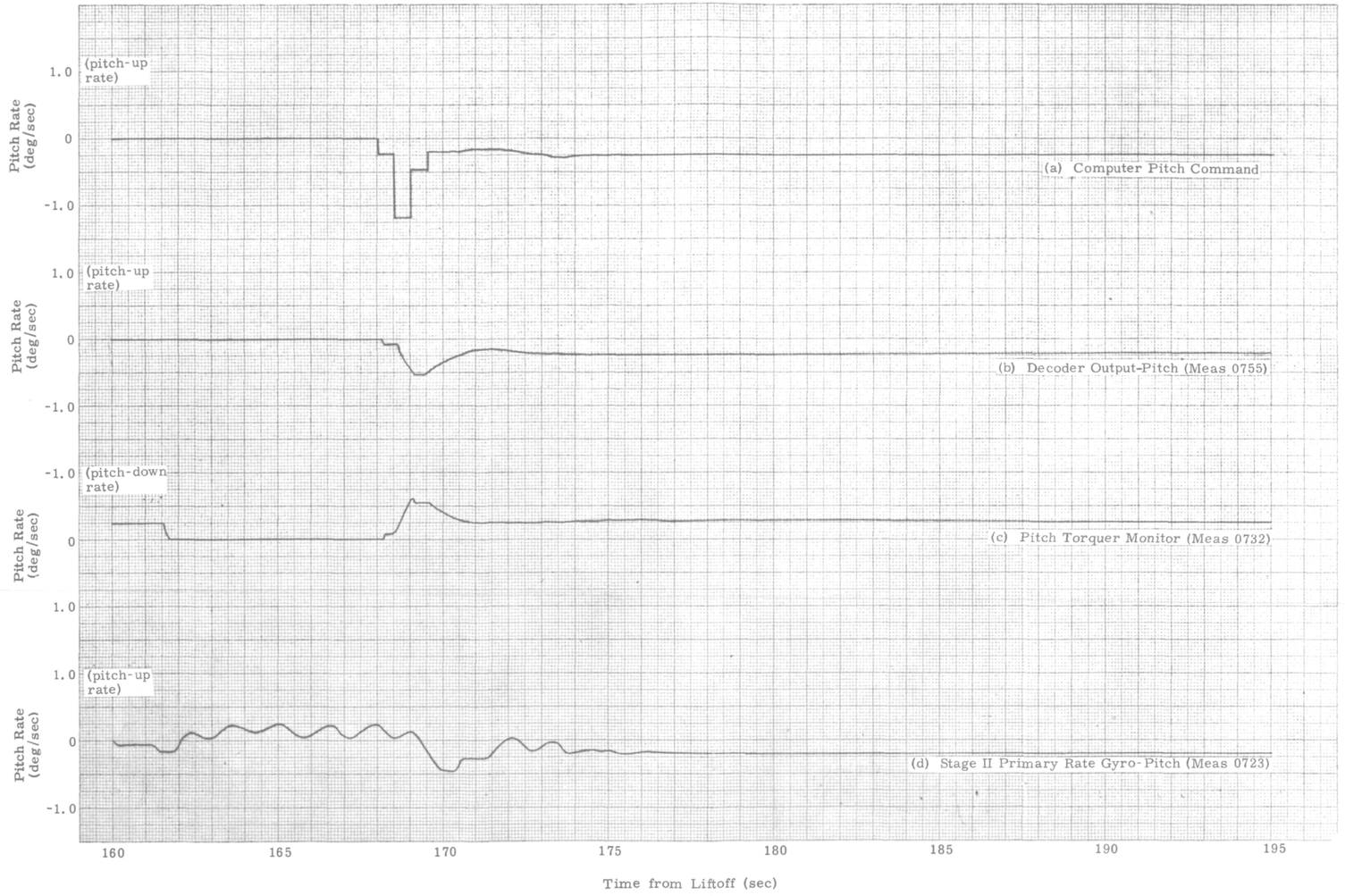


Fig. VI-1. GT-6A Stage II RGS Pitch Guidance Flight History

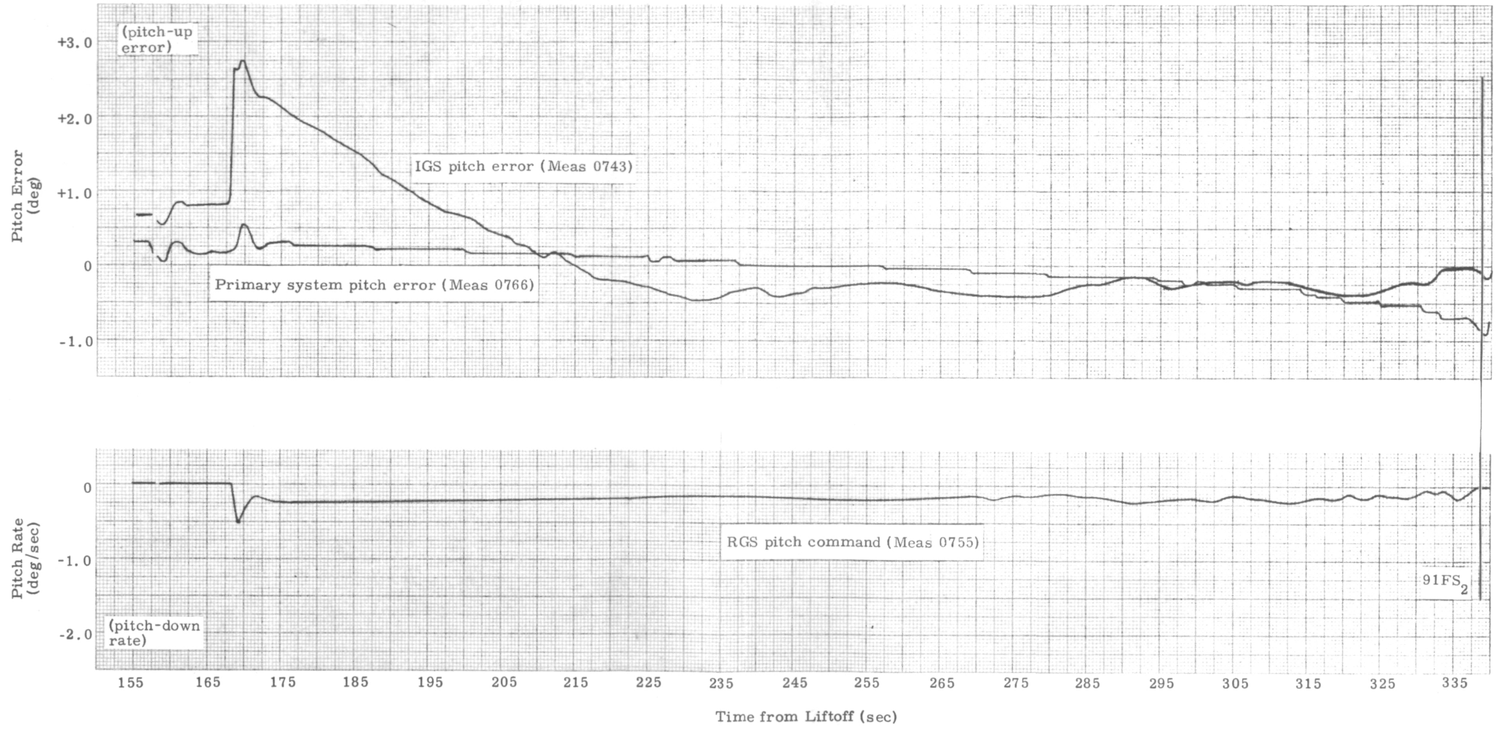


Fig. VI-2. Stage II IGS Pitch Flight History

this, a 0.56 deg/sec pitch-down command was issued for approximately 0.3 second. Thereafter, the pitch steering command decreased to 0.25 deg/sec within two seconds. Throughout the remainder of flight, the pitch commands remained between 0.05 and 0.25 deg/sec pitch-down. In the latter portion of Stage II flight, very small magnitude oscillations built up and became noticeable at about SECO - 60 seconds, and lasted until SECO. These oscillations, which are attributed to atmospheric noise effects, were approximately the same amplitude as those observed on the flight of GT-7 and were much smaller than those observed on GT-5 flight.

b. Yaw steering

Decoder yaw steering commands began at LO + 169.23 seconds. The command peaked at 0.20 deg/sec, yaw right, at approximately LO + 180 seconds, and then decreased to 0.025 deg/sec, yaw right, by LO + 215 seconds. The yaw commands remained between 0 and 0.04 deg/sec, yaw right, for the remainder of the Stage II flight period. The decoder yaw steering output is shown in Fig. VI-3 for the entire Stage II flight.

c. Discrete commands

The times for the computer-generated SECO/ASCO command and the vehicle reactions are shown in Table VI-2.

TABLE VI-2

SECO/ASCO Events

Signal	Meas	Time from Liftoff (ms)
Ground station SECO/ASCO	--	338.681 ± 3
Decoder discrete output	0777	338.717 ± 5
$91FS_2$	0519	338.737 ± 5
ASCO	0799	338.770 ± 25

The data shown in Table VI-2 indicate that the SECO time delay from ground station issuance to $91FS_2$ was 56 ± 8 milliseconds. The time delay between $91FS_2$ and ASCO reception was 33 ± 30 milliseconds.

5. GT-6 and GT-6A Launch Attempts

There were no anomalies in radio guidance system performance during either the GT-6 25 October 1965 launch countdown or the GT-6A 12 December 1965 launch countdown and launch attempt.

[redacted]

II-6-2

[redacted]

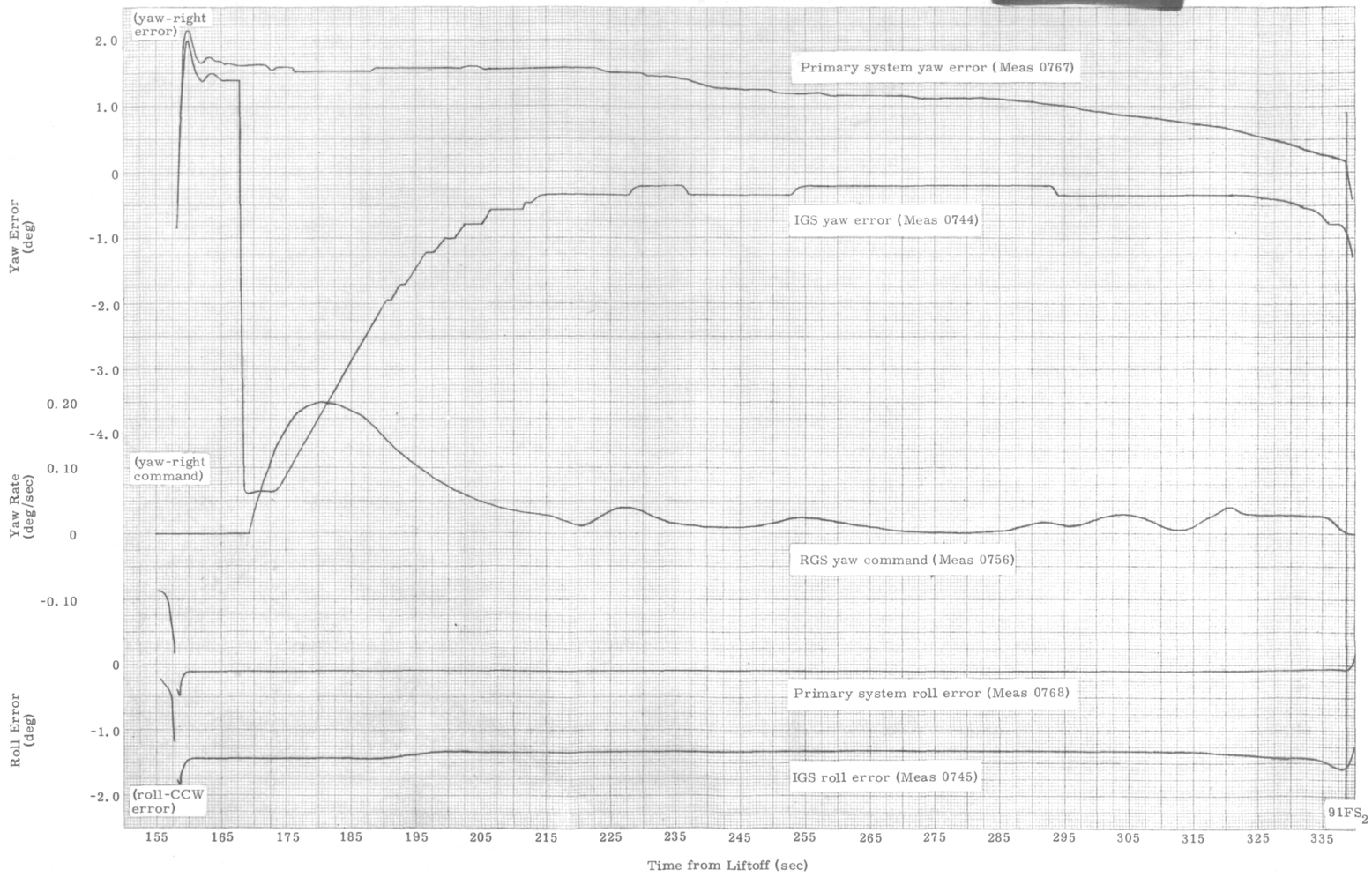


Fig. VI-3. Stage II IGS Yaw/Roll Guidance Flight History

B. SPACECRAFT INERTIAL GUIDANCE SYSTEM ASCENT PERFORMANCE

1. Prelaunch Nulls

The prelaunch IGS attitude error null signals were as follows:

Pitch	-0.185 degree
Yaw	-0.249 degree
Roll	-0.064 degree

These null signals were well within the specification values of ± 0.37 degree in pitch and yaw, and ± 0.25 degree in roll.

2. Stage I Performance

IGS performance during Stage I flight correlated well with the primary system, as shown by a comparison of IGS and corresponding primary system attitude errors in Figs. IV-2 through IV-4. The BECO dispersion between IGS and primary system attitude errors is discussed in Chapter IV.

The IGS Stage I gain change discrete was issued at LO + 109.829 seconds ± 0.025 second, which was well within the specification time of 110.00 seconds ± 1%.

3. Stage II Performance

IGS pitch, yaw and roll performance during Stage II flight was normal. The attitude error dispersions which had built up between the IGS and primary system during Stage I flight in pitch, yaw and roll were apparent in the early portion of Stage II flight as shown in Figs. VI-2 and VI-3.

a. Stage II pitch

IGS Stage II pitch attitude error appears in Fig. VI-2. Primary system pitch attitude error and RGS pitch steering commands are shown for comparison.

IGS closed-loop pitch guidance began at LO + 168.13 seconds. IGS pitch attitude error peaked at + 2.75 degrees shortly thereafter and remained there for approximately 0.45 second. Figure VI-2 shows that, due to the RGS pitch rate command, the TARS pitch attitude error builds up during this same time period to about +0.55 degree. IGS pitch behavior during this period was normal and compares well with primary system behavior in correcting the vehicle trajectory errors.

IGS pitch attitude error decreased to null within 45 seconds as the RGS pitched the vehicle down. For the remainder of Stage II flight, IGS pitch remained within limits of 0.0 to -0.5 degree.

b. Stage II yaw

IGS Stage II yaw attitude error is shown in Fig. VI-3. Primary system yaw attitude error and RGS yaw steering commands are shown for comparison.

IGS yaw performance throughout Stage II powered flight appeared normal. Steering began approximately at the same time as pitch. IGS yaw attitude error was approximately +1.4 degrees just prior to initiation of closed-loop steering. Shortly after steering began, IGS yaw attitude error peaked at approximately -4.9 degrees and remained at this level for about five seconds. The direction and magnitude of the IGS yaw attitude error change at guidance initiation is approximately the value that would be predicted due to the effects of biased azimuth steering. Thereafter, the IGS yaw attitude error decreased to -0.35 degree in approximately 40 seconds as the RGS command yawed the vehicle right. Subsequently, the IGS yaw attitude error remained within approximately -0.20 to -0.35 degree of null until about LO + 320 seconds. At this time, the IGS yaw attitude error began to slope in the negative direction, and by SECO it had built up to about -0.95 degree, which would be a GLV yaw-right command. The amplitude of the attitude error was not excessive and the direction of the attitude error buildup was as expected due to center of gravity drift. In Fig. VI-3, a similar effect is apparent in primary attitude error in that the error is building up negatively and also in the small RGS yaw steering command, which commands the GLV to yaw right.

c. Stage II roll

IGS roll attitude error for Stage II is shown in Fig. VI-3, with TARS roll attitude error shown for comparison. There was a small apparent drift rate between TARS and IGS roll as shown by the small increase in IGS roll output between approximately LO + 310 seconds and SECO. The drift rate was CCW, IGS with respect to TARS, and the buildup in IGS error between the referenced times was about -0.25 degree. The dispersion is predominantly due to TARS roll gyro g-sensitive drift; this type of dispersion has been noted on all flights to date.

4. GT-6 and GT-6A Launch Attempts

There were no anomalies noted in IGS attitude error data during either the GT-6 25 October 1965 launch countdown or the GT-6A 12 December 1965 launch countdown and launch attempt.

VII. ELECTRICAL SYSTEM ANALYSIS

A. CONFIGURATION

The launch vehicle airborne electrical system components installed for the GT-6A flight were similar to those used on GLV-5 and GLV-7 except for the flashing beacon light assembly, which was a special GLV-7 feature.

The launch attempt countdown of 12 December progressed through power transfer to airborne batteries and into engine ignition. Since the airborne batteries had experienced partial discharge, they were replaced for the successful launch and flight of 15 December.

B. COUNTDOWN AND FLIGHT PERFORMANCE

The airborne electrical system functioned as designed during the launch attempt of 12 December and through the entire flight of 15 December, with all parameters within specifications. Liftoff on 15 December occurred without incident.

During staging, the IPS current trace indicated that one separation nut on the Stage I side shorted to structure, maintaining a current increase of 18.7 amperes for approximately 0.4 second. Comparable staging shorts were encountered on GT-1, GT-2 and GT-5 flights, and no staging shorts were encountered on GT-3, GT-4 and GT-7 flights.

Currents to the Stage II redundant shutdown squibs at SECO were not detectable on either the APS or IPS traces, although squib operation was confirmed by Meas 0521.

At spacecraft separation, the launch vehicle/spacecraft electrical interface was cut by a guillotine in the adapter. This caused a 21-ampere rise in APS bus current for approximately 100 milliseconds, indicating a momentary shorting of GLV interface signals to structure. No similar increase was noted on the IPS bus. Guillotine shorts are expected and have occurred on all spacecraft separations to date; they have created no detrimental effects.

A summary of electrical system parameters measured at power transfer and at liftoff signal for the launch attempt of 12 December is presented in Table VII-1. A summary of these parameters measured at power transfer and during flight for the launch of 15 December is presented in Table VII-2.

TABLE VII-1

GLV-6 Launch Attempt Measurements

Meas	Parameter	At Power Transfer			At Liftoff
		Before	During	After	
0800	IPS (volts)	29.8	27.5	29.1	29.2
0804	IPS (amp)	28.4	26.4	27.9	26.0
0801	APS (volts)	30.0	29.8	30.2	29.8
0805	APS (amp)	24.0	24.0	24.0	24.0
0802	400 cycle, Phase A (volts)	113.9	113.9	113.9	113.8
0803	400 cycle, Phase A (cps)	400.8	400.8	400.8	400.8
0726	25-vdc power supply (volts)	25.3	25.3	25.3	25.3

TABLE VII-2

GLV-6 Launch Measurements

Meas	Parameter	At Power Transfer				At $87FS_2/91FS_1$ (staging)			At Spacecraft Separation		
		Before	During	After	At Liftoff	Before	During	After	Before	Peak	After
0800	IPS (volts)	29.8	27.4	29.1	29.1	29.1	28.0	29.4	29.7	29.7	29.7
0804	IPS (amp)	26.9	25.9	26.9	27.8	26.6	45.3	26.6	23.1	23.1	23.1
0801	APS (volts)	29.7	29.7	29.7	29.6	29.5	29.5	29.6	29.8	29.0	29.8
0805*	APS (amp)	27.3	27.3	24.2	24.0	28.3	31.3	25.2	25.8	46.6	22.1
0802	400 cycle, Phase A (volts)	113.8	113.7	113.8	113.7	113.7	113.7	113.7	113.7	113.7	113.7
0803	400 cycle, Phase A (cps)	400.7	400.7	400.7	400.6	400.5	400.5	400.5	400.7	400.7	400.7
0726	25-vdc power supply (volts)	25.3	25.3	25.3	25.3	25.3	25.2	25.3	25.3	25.3	25.3

*Including TARS heater cycling current.

VIII. INSTRUMENTATION SYSTEM

A. AIRBORNE INSTRUMENTATION

1. Prelaunch and Countdown Status

The airborne instrumentation system operated within specified limits during the GT-6A launch attempt and during the GT-6A prelaunch testing and countdown. No components in the system were replaced after the simulated flight test.

Ambient checks during the launch countdown verified that all measurements were operating prior to T-0.

2. Data Acquisition

The measurements program for this launch consisted of 149 PCM analog signals and 42 PCM bilevel signals. All channels functioned throughtout flight.

S/A 3 oxidizer pump discharge pressure (Meas 0509) displayed approximately a 3% oscillation around the ambient level during Stage I flight. After Stage II ignition, the data appeared normal.

Actuator 4_1 position Meas 0153 at T-21 seconds started drifting off null. At liftoff, the measurement returned to its normal position and functioned normally for the balance of the flight.

3. Instrumentation System Parameters

Instrumentation system parameters, as measured in flight, are compared with specified limits in Table VIII-1. All data were within the required limits.

4. Telemetry Signal Strength (244.3 mc)

Telemetry signal strength records indicated satisfactory signal levels for data acquisition from the launch vehicle from liftoff to approximately SECO + 114 seconds. The anticipated staging blackout lasted approximately 300 milliseconds.

A drop in telemetry signal strength of approximately 15 db occurred at LO + 289.3 seconds for a period of 2.5 seconds; as recorded on signal strength records from Tel II, Grand Bahama Island and Grand Turk station, however, there was no loss of data on any programmed measurement, and the signal level recovered to the normal level following the 2.5-second interval.

TABLE VIII-1

Instrumentation System Parameters

Meas	Description	Required PCM Count / Required Engineering Units	PCM Count / Engineering Units: At Liftoff	At 87FS$_2$/ 91FS$_1$	At 91FS$_2$	At SECO + 20
0810	5-volt power supply	203 ± 1 / 5.00 V ± 0.035	203 / 5.001	203 / 5.001	203 / 5.001	203 / 5.001
0811	40-volt power supply	229 ± 3 / 40.00 V ± 0.64-0.44	229 / 40.10	229-230 / 40.10-40.27	230 / 40.27	230 / 40.27
0813	PCM mercury cell	73 ± 1 / 1.335 to 1.375 V	72 / 1.335	72 / 1.335	72 / 1.335	73 / 1.355
0814	PCM mercury cell	73 ± 1 / 1.335 to 1.375 V	72 / 1.335	72 / 1.335	72 / 1.335	72 / 1.335
0815	PCM mercury cell	73 ± 1 / 1.335 to 1.375 V	74 / 1.375	74 / 1.375	74 / 1.375	74 / 1.375
0816	+30-volt power supply	168 ± 3 / 29.92 to 30.06 V	169 / 30.03	168-169 / 30.01-30.03	168-169 / 30.01-30.03	168-169 / 30.01-30.03
0817	-30-volt power supply	185 ± 3 / 29.92 to -30.08 V	186 / -30.03	186 / -30.03	186 / -30.03	186 / -30.03

The Cape Kennedy Tel II and Tel III ground stations monitored the entire flight of the launch vehicle. The Grand Bahama Island station acquired data from approximately LO + 40 seconds to the end of flight. The Grand Turk station acquired data during Stage II flight, beginning at approximately LO + 181 seconds.

B. LANDLINE INSTRUMENTATION

1. Countdown Status

The landline instrumentation system operated satisfactorily from the start of propellant conditioning through the launch countdown up to lift-off. All instrumentation holdfire functions monitored in the blockhouse remained within specification throughout the countdown.

2. Data Acquisition

During propellant loading and launch countdown for GT-6A, a total of 133 measurements consisting of 55 landline measurements on strip-chart recorders, 23 landline measurements on magnetic tape, and 55 airborne measurements in real-time on chart recorders were monitored and recorded.

Data acquisition for all landline monitored data was 100%.

IX. RANGE SAFETY AND ORDNANCE

A. COMMAND CONTROL RECEIVERS

1. Countdown Performance

The command receiver shutdown and destruct test and the ASCO test were successfully completed. Telemetry indicated a stable signal strength level of greater than 160 microvolts from T-3 minutes through liftoff.

2. Flight Performance

Command receivers S/N 33(APS) and S/N 36(IPS) were flown on GLV-6. The RF carrier level increased to about 1400 microvolts at LO + 13.5 seconds, fluctuated between 500 and 1200 microvolts until LO + 60 seconds, at which time the carrier dipped sharply to 2.6 microvolts. The total excursion during the carrier dip was approximately 5 seconds. At LO + 67 seconds, transmissions from Cape Kennedy were switched from low power to high power, and at LO + 73 seconds the signal level reached approximately 2400 microvolts, maximum for the flight. At LO + 119 seconds, the RF carrier was transferred from Station 1 to Station 3 (Grand Bahama Island). The RF carrier level dropped to a range from 100 to 165 microvolts and remained at this level throughout the Grand Bahama pass. At LO + 259 seconds, the RF carrier was transferred from Station 3 to Station 7 (Grank Turk), with no change in signal level at transfer. The RF carrier level remained between 100 and 160 microvolts until LO + 370.5 seconds, at which time the carrier was transferred from Station 7 to Station 3 and the level dropped to about 8 microvolts. At LO + 393 seconds, the RF carrier again was transferred to Station 7, and the level increased to a range from 60 to 120 microvolts. Signal level at the time ASCO was transmitted was 135 microvolts, and the time between SECO and ASCO was about 50 milliseconds.

B. MISTRAM

1. Countdown Performance

The MISTRAM open-loop checks with the MACK station were completed successfully. Telemetry indicated that the transponder was locked onto the MACK station from T-3 minutes until LO + 1.6 seconds, when the MACK station signal was manually removed.

2. Flight Performance

Airborne transponder. Transponder S/N 116 was flown on GLV-6. Telemetry data indicated that performance was very good. MISTRAM I (Valkaria) acquired and locked on both channels at LO + 5.6 seconds and started active track at LO + 24.7 seconds. During the first 100 seconds of flight, there were two calibration channel unlocks and several sweep changes indicative of poor received data at the ground station. There was no loss of reconstructable data during this period. Yaw attitude disturbances due to winds aloft and yaw guidance commands have been correlated with calibration channel unlock and sweep changes. The transponder did not unlock when the staging power transient occurred.

3. MISTRAM I Station (Valkaria)

The Valkaria station obtained reconstructable data from LO + 24 seconds until LO + 63 seconds, LO + 64.5 seconds to LO + 158 seconds, and LO + 168 seconds to LO + 385 seconds. Over the programmed prime use of MISTRAM data for impact prediction, LO + 60 seconds to spacecraft separation, the Valkaria MISTRAM data were used for a total of 266.7 seconds, or 88.6 % of the time. Utilization of data for the primary and secondary impact prediction plots is included in Table IX-1.

4. MISTRAM II Station (Eleuthera)

The Eleuthera station acquired the vehicle at handover, LO + 388 seconds, at elevation angles less than 0.5 degree. Three active sweeps were obtained before the station lost the signal. MISTRAM data from Station II were not used for impact prediction.

C. ORDNANCE

The prevalves used during the GT-6A launch attempt on 12 December were not replaced for the GT-6A launch on 15 December; therefore, actuated prevalve pressure cartridges were left electrically connected. The Stage I engine start cartridges and dropweight ordnance operated satisfactorily. All four holddown bolts and all lower launch nuts were recovered after the launch, indicating proper operation of all launch release ordnance.

Stage separation pressure cartridges and explosive nuts and the Stage II start cartridge operated as required. The TARS staging arm timer signal occurred at LO + 144.41 seconds and the IPS staging arm timer signal occurred at LO + 144.97 seconds. Both times are compatible with GLV-6 trend data.

TABLE IX-1

Range Safety Plotboards Impact Prediction

Primary Plotboard		Secondary Plotboard	
System	Usage Time (sec)	System	Usage Time (sec)
MISTRAM I	300.6	Mod III	330.1
Patrick AFB TPQ-18	32.4	Patrick AFB TPQ-18	14.1
Merritt Island TPQ-18	5.1	Merritt Island TPQ-18	33.3
Cape Kennedy FPS-16	56.3	Cape Kennedy FPS-16	3.6
Grand Bahama TPQ-18	3.6	Grand Bahama TPQ-18	5.7
Grand Turk TPQ-18	1.0	Grand Turk TPQ-18	16.5
Bermuda FPS-16	8.7	Bermuda FPS-16	4.4
TOTAL	407.7	TOTAL	407.7

X. MALFUNCTION DETECTION SYSTEM

A. CONFIGURATION

The malfunction detection system (MDS) hardware installed on GLV-6 for the launch attempt countdown on 25 October 1965, the launch attempt on 12 December 1965, and the flight on 15 December 1965 is presented in Table X-1.

There were no MDS hardware changes from the time of the GT-6 launch attempt countdown until the flight of GT-6A.

TABLE X-1

MDS Components

Nomenclature	Part Number	Manufacturer	Serial Number
Rate switch package	PS830600015-027	Giannini	4010
Malfunction detection package	424-7569205-189	Martin	B001
Tank pressure transducers, Stage I	PS746000002-023	Servonics	Fuel A, 1065 Fuel B, 1068 Oxidizer A, 1097 Oxidizer B, 1078
Tank pressure transducers, Stage II	PS746000002-025	Servonics	Fuel A, 2101 Fuel B, 2075 Oxidizer A, 2107 Oxidizer B, 2108
Stage separation connectors	CCI8119A1-5 CCI8119A1-6	Cannon	00110 00104
MDS engine switches, Stage I	284321	Aerojet	S/A 1 primary, 0000559 S/A 1 redundant, 0000007 S/A 2 primary, 0000035 S/A 2 redundant, 0000726
MDS engine switches, Stage II	711049-1	Aerojet	S/A 3 primary, 0000837 S/A 3 redundant, 0000569

B. SYSTEM PERFORMANCE

Performance of the MDS during the countdowns, launch attempt and flight of GT-6A was satisfactory.

1. Engine Pressure Switches

Operation of the Stage I engine malfunction detection thrust chamber pressure switches (MDTCPS) and the Stage II engine malfunction detection fuel injector pressure switches (MDFJPS) for the launch attempt and the flight is summarized in Table X-2 and Table X-3, respectively. These switches are required to "make" in a pressure range of 540 to 600 psia and "break" in a pressure range of 585 to 515 psia. During both the launch attempt and the flight, the Stage I engine start transient was of sufficient amplitude and time duration to cause the S/A 2 MDTCPS switches to respond momentarily to the thrust chamber pressure. All MDS engine pressure switches operated properly and within specification requirements.

TABLE X-2

GT-6A Launch Attempt Operation of MDS Engine Pressure Switches

	S/A 1 (Meas 0356)	S/A 2 (Meas 0357)	S/A 3 (Meas 0855)
Make (540 to 600 psia)	1454:04.222 at 600 psia	1454:04.312 at 580 psia	N/A*
Break (585 to 515 psia)	1454:04.775 at 545 psia	1454:04.472 at 550 psia	N/A*

* Not applicable; the Stage II engines were not started.

TABLE X-3

GT-6A Launch Operation of MDS Engine Pressure Switches

	S/A 1 (Meas 0356)	S/A 2 (Meas 0357)	S/A 3 (Meas 0855)
Make (540 to 600 psia)	1337:24.092 at 585 psia	1337:24.112 at 575 psia	1340:04.370*
Break (585 to 515 psia)	1340:03.586 at 550 psia	1340:03.588 at 530 psia	1343:05.348*

* S/A 3 fuel injector pressure is not instrumented on the Gemini Launch Vehicle; hence, make and break pressures were unavailable.

2. Switchover

The MDS switchover circuitry functioned properly throughout the flight. There were no switchover commands and no switchover was executed--indicating proper performance of the switchover circuitry.

3. Vehicle Rate Detection

The spin motor rotation detectors (SMRDs) contained in the malfunction detection package functioned properly. The SMRDs monitor rate switch package (RSP) gyro rotational speed and thereby its rate sensing capability.

The rate switch package operated properly throughout the countdowns and flight. There were no vehicle overrates detected by the MDS, and none occurred during flight from liftoff through SECO + 20 seconds. Table X-4 compares the maximum launch vehicle rates, measured during the period from liftoff through SECO, with the RSP switch settings.

TABLE X-4

Maximum Vehicle Rate Compared with Rate Switch Settings

	Axes	Stage I Flight	Flight Event	Stage II Flight	Flight Event
Rate switch settings (deg/sec)	Pitch	+ 2.5: -3.0	N/A	± 10	N/A
	Yaw	± 2.5	N/A	± 10	N/A
	Roll	± 20.0	N/A	± 20	N/A
Maximum vehicle rates (deg/sec)	Pitch	- 1.12	Wind Shear	+ 2.36	Staging
	Yaw	- 1.41	Wind Shear	+ 2.62	Staging
	Roll	+ 1.90	Roll Program	- 3.67	Staging

During Stage II flight there was a closing and opening of the yaw A low-rate switch in response to the vehicle rates. A switchover signal was not initiated since at stage separation the rate switch settings that must be exceeded to generate an overrate and switchover signal are changed from the Stage I values to the higher Stage II values as shown in Table X-4.

Following SECO + 23 seconds (after spacecraft separation), there were ten operations of the rate switches. The rate gyro outputs verified that the rate switch performance was in agreement with the RSP calibration data. Table X-5 summarizes the rate switch operations.

TABLE X-5

Rate Switch Operation

Specification Switch Operation Limits (deg/sec)	RSP Calibration Data		Time of Rate Switch Operation	Rate Gyro Output	
	Primary (deg/sec)	Redundant (deg/sec)		Primary (deg/sec)	Redundant (deg/sec)
Yaw close "A" switch (2.08 to 2.92)	2.50	N/A	Stage separation + 0.885 sec	2.28	2.46
Yaw open "A" switch (2.50 to 1.88)	2.26	N/A	Stage separation + 1.035 sec*	1.74	2.08
Yaw close "A" switch (2.08 to 2.92)	2.50	N/A	SECO + 31.362 sec	2.23	2.46
Yaw close "B" switch (2.08 to 2.92)	N/A	2.47	SECO + 32.512 sec	2.48	2.56
Yaw open "B" switch (2.47 to 1.85)	N/A	2.29	SECO + 36.162 sec	2.08	2.41
Yaw open "A" switch (2.50 to 1.88)	2.26	N/A	SECO + 36.562 sec*	2.04	2.36
Pitch close "B" switch (2.08 to 2.92)	N/A	2.38	SECO + 48.812 sec	2.23	2.52
Pitch close "A" switch (2.08 to 2.92)	2.47	N/A	SECO + 48.932 sec	2.28	2.53
Pitch open "A" switch (2.47 to 1.85)	2.16	N/A	SECO + 48.962 sec*	2.18	2.43
Pitch open "B" switch (2.38 to 1.78)	N/A	2.24	SECO + 49.012 sec	2.18	2.43
Pitch close "A" & "B" switches (2.08 to 2.92)	2.47	2.38	SECO + 49.562 sec	2.28	2.53

* Time is 0.40 second less than the TLM indication. This is due to a 0.40-second R-C time delay associated with the monitored circuits.

4. Tank Pressure Sensors

All MDS tank pressure transducers operated properly throughout the countdowns and the flight. The maximum difference between the transducer pairs on each tank is presented in Table X-6.

TABLE X-6

Maximum Voltage and Pressure Differences
Between Tank Pressure Transducer Pairs

	Maximum Difference			
	Δ Volts (telemetry)	Percent of Transducer Full Range	Δ psi	Percent of Transducer Full Range
Stage I fuel	0.040	0.80	0.35	0.70
Stage I oxidizer	0.060	1.20	0.28	0.56
Stage II fuel	0.090	1.80	0.78	1.04
Stage II oxidizer	0.060	1.20	0.87	1.16

Figure X-1 presents the calibration curves for the Stage I fuel tank pressure transducer pairs (A and B) to clarify the percentage variations between voltages and psi (shown in Table X-6). The maximum difference of 1.80% of transducer full-range output voltage is well within the transducer and telemetry system errors.

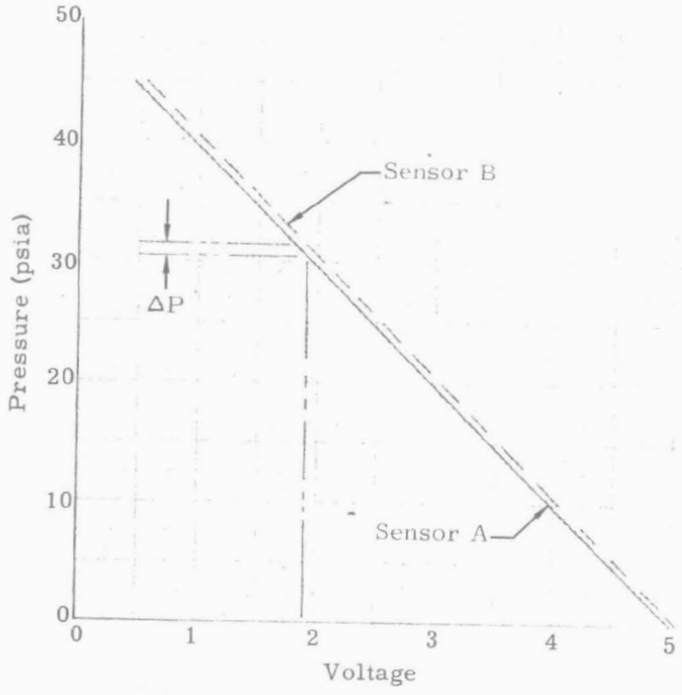


Fig. X-1. Calibration Curves for Stage I Fuel Tank Pressure Transducers

XI. CREW SAFETY

A. GT-6A LAUNCH ATTEMPT

On the GT-6A launch attempt, the astronauts received a series of cues during the pad shutdown event which severely tested the effectivity of their abort training. Figure XI-1 shows: (a) a normal start and liftoff sequence, (b) the sequence of events which occurred during the GT-6A pad shutdown and (c) a pad fallback event which would require immediate abort action. As noted in Fig. XI-1 (a), the Stage I engine lights are normally on until engine chamber pressure exceeds 68% of nominal at which time the S/A 1 and S/A 2 engine lights go out and remain out until BECO. Trajectory liftoff normally occurs 2.1 seconds after the engine lights go out (approximately 0.25 g increase in acceleration), and the spacecraft event timer starts running approximately 0.2 second after trajectory liftoff. The launch nut fire signal is sent 0.03 second prior to trajectory liftoff, and at this signal CAPCOM announces to the astronauts, "Liftoff."

Figure XI-1 (b) depicts the GT-6A launch attempt events. The astronauts felt the vibrations and heard the audible backup cues of engine start and shutdown. The engine light was observed to wink out and then on again.* However, even though an event timer start was noted, the required second liftoff cue of a verbal comment from CAPCOM indicating liftoff did not occur. Therefore, the 2.3-second early start of the event timer was interpreted (correctly) to be a false signal and no abort action was taken.

For comparison purposes only, a pad fallback event is depicted in Fig. XI-1 (c). It should be noted that this abort event is quite different from the GT-6A shutdown.

B. PRELAUNCH WINDS OPERATIONS

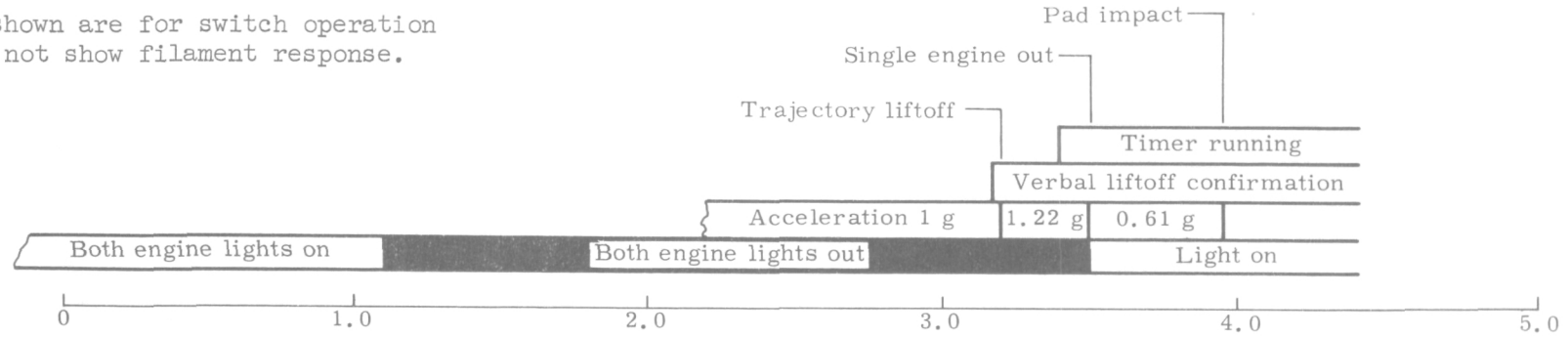
All wind profiles for the GT-6A launch attempt and the GT-6A launch were mild quartering tail winds with light shears through the high dynamic pressure period of the trajectory. All profiles were well within the design specification. The vehicle load simulations, using the measured profiles, predicted peak vehicle loads well within the launch winds load criteria.

The mild winds and low predicted peak flight loads resulted in a launch winds go status throughout the wind surveillance period for both the launch attempt and launch.

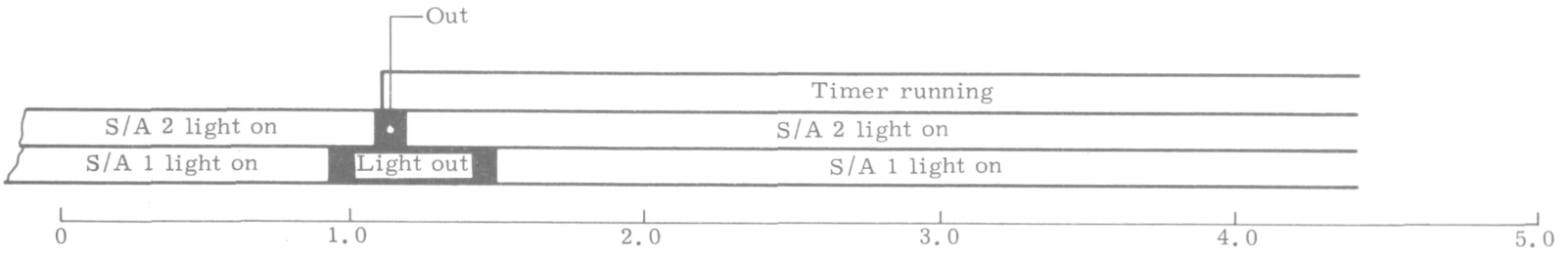
The prelaunch winds operations ran smoothly except for one computer and one communication problem prior to the GT-6A launch. The

*Only one engine light was observed to wink out.

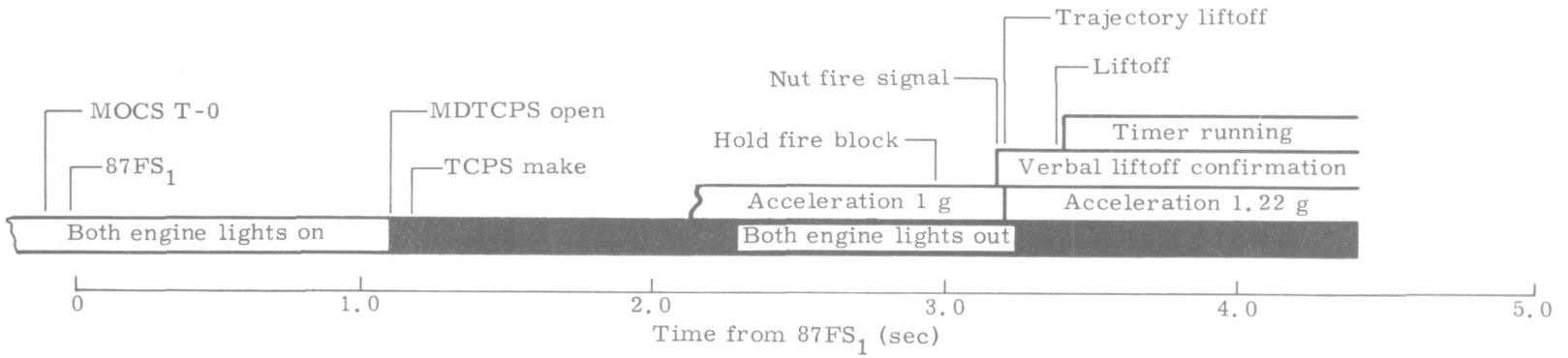
NOTE: Times shown are for switch operation and do not show filament response.



(c) Pad Fallback Event



(b) GT-6A Launch Attempt



(a) Normal Launch and Liftoff

Fig. XI-1. Sequence of Events During Launch

7094 digital trajectory program failed while trying to obtain a check run using the F-1 day wind data. This card reading problem was corrected prior to the flight day operations. The F-2 day and F-1 day Datafax messages to Houston were delayed until late on F-1 day because of a Datafax failure at Houston. The major details of these messages were telephoned on schedule, and the problem was corrected for the flight day operations.

The prelaunch wind profiles released by the Air Weather Service (AWS) for the GT-6A launch attempt and the GT-6A launch are presented in Figs. XI-2 through XI-7. The T-0 hour profile (Figs. XI-2 and XI-7) was not used in the launch winds program and is shown for reference only. The wind profiles are summarized in Tables XI-1 and XI-2 which indicate run number, time of the sounding release, operations performed, messages sent, ratio of peak vehicle load to limit load, a wind in or out of specification statement and the launch winds go or no-go status.

Figure XI-8 presents a block diagram of the prelaunch wind operations and indicates the data flow between Cape Kennedy, Martin-Baltimore and MCC-NASA, Houston.

TABLE XI-1

Summary of Prelaunch Operations--GT-6A Launch Attempt
(12 December 1965)

Run No.	Time of Data Release to Martin-Baltimore	Operation
1	F-2 days 1100 EST 12-10-65	Wind comparison to specification. Sent to Cape Kennedy and MCC-Houston. Wind in specification; status go.
2	F-1 day 1100 EST 12-11-65	Wind comparison to specification and trajectory simulation. Sent to Cape Kennedy and MCC-Houston. Wind in specification; status go.
3	T-7 hours 0250 EST 12-12-65	Wind comparison to specification, trajectory, vehicle loads, vehicle response, tank underpressure constraints, switchover load constraints and switchover temperature constraint were determined by computer programs. Data sent to Cape Kennedy and MCC-Houston. Wind in specification; load ratio 0.85; status go.

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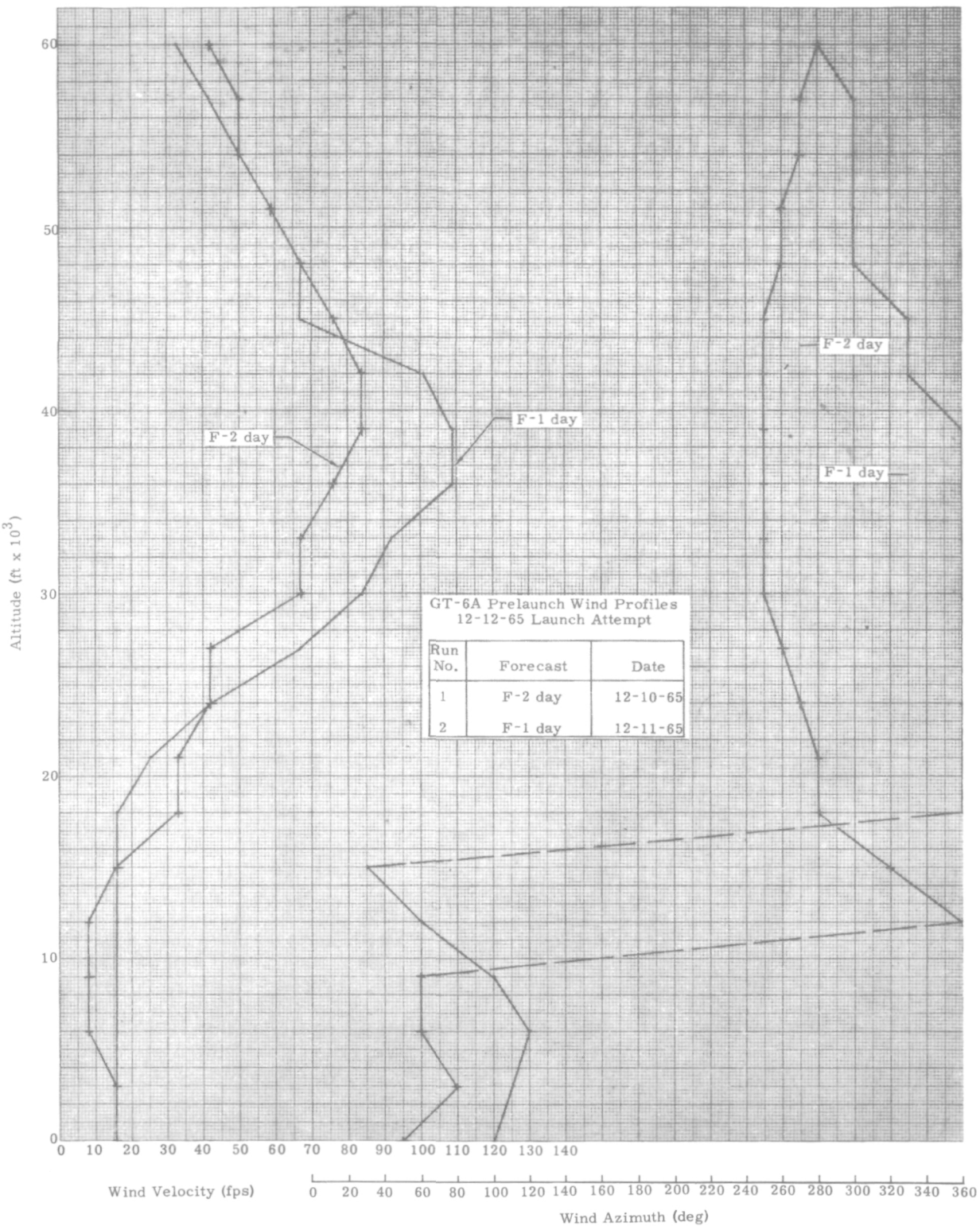

Fig. XI-2. Launch Attempt F-2 and F-1 Day Wind Profiles

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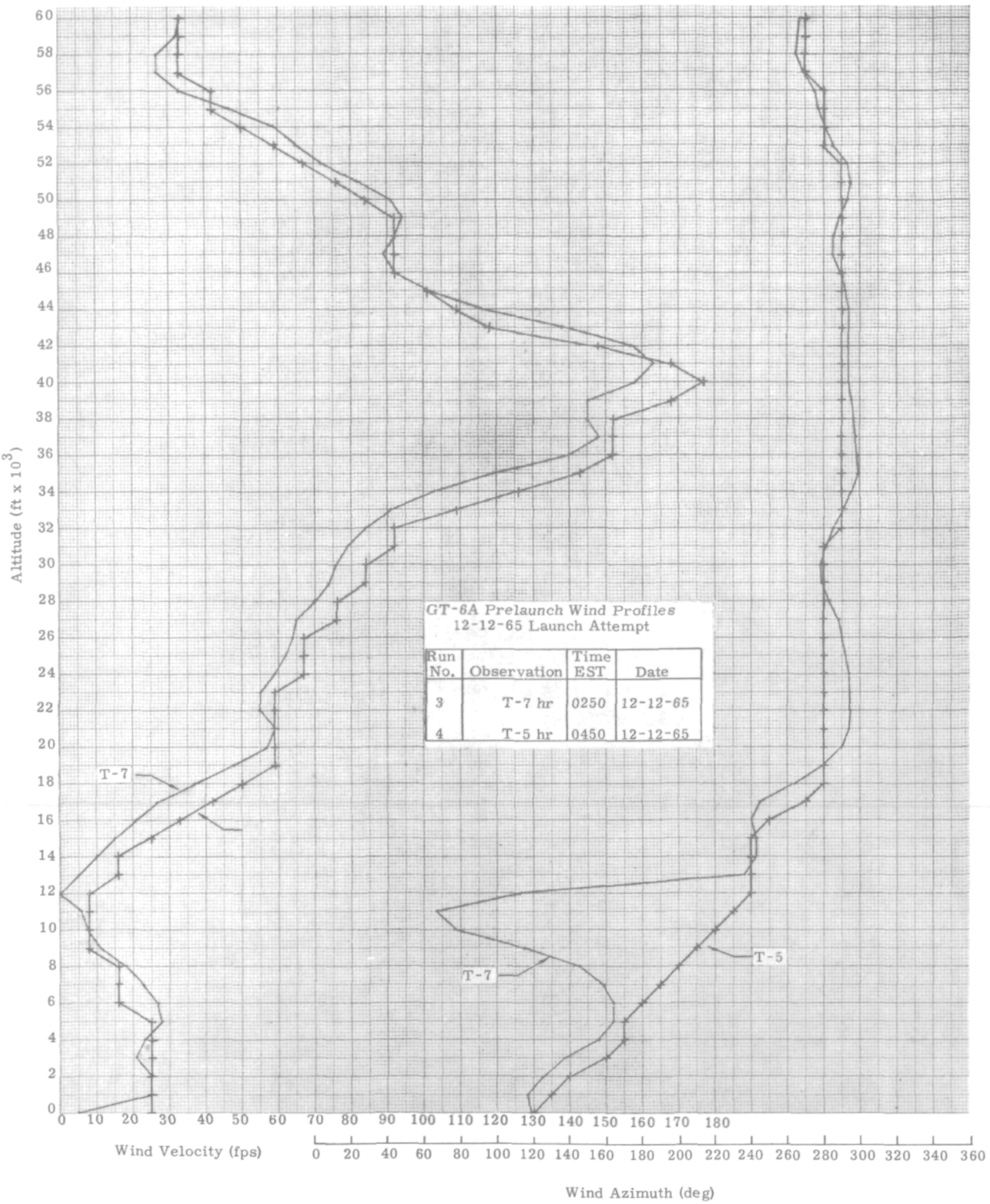


Fig. XI-3. Launch Attempt T-7 and T-5 Hour Wind Profiles

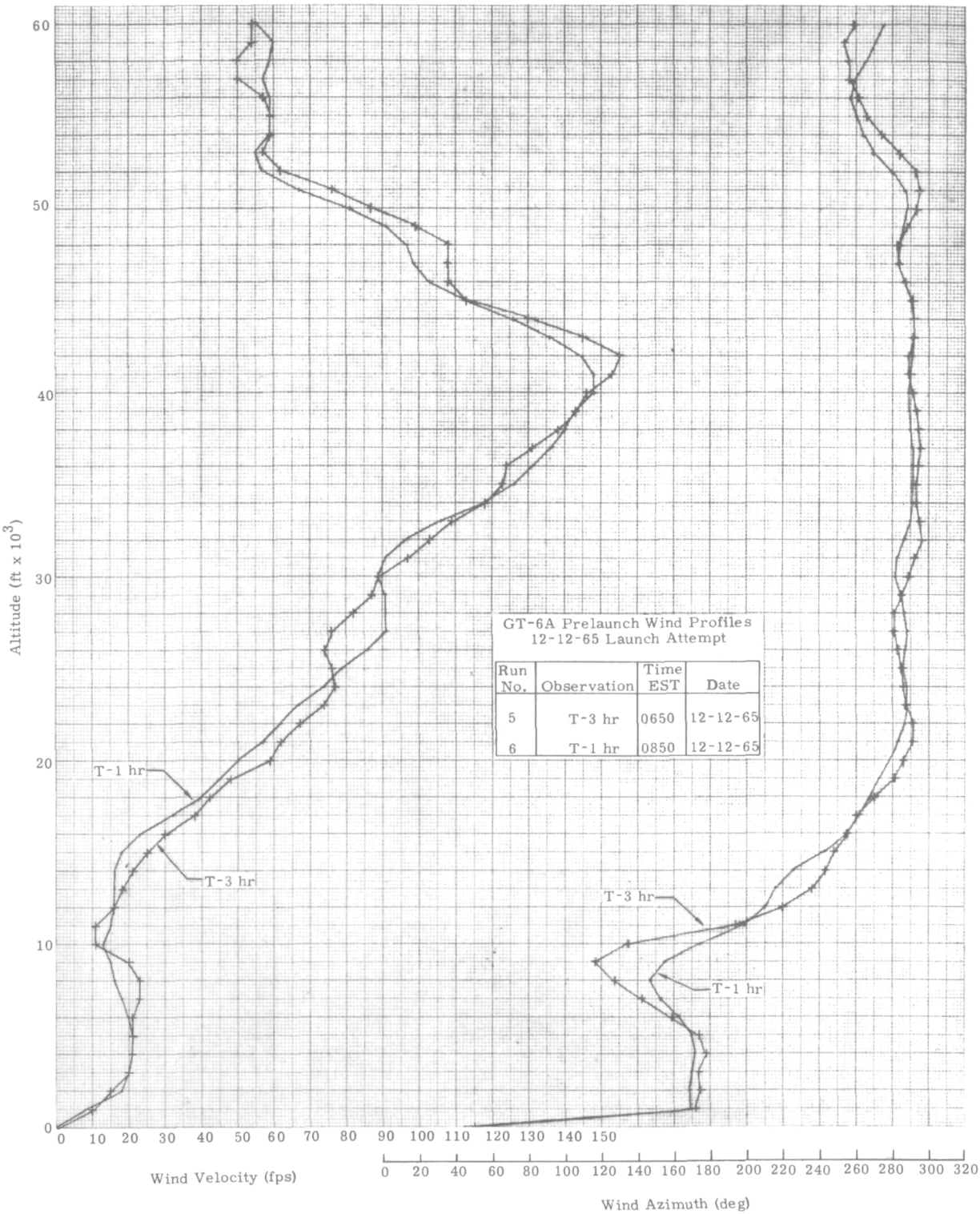


Fig. XI-4. Launch Attempt T-4 and T-3 Hour Wind Profiles

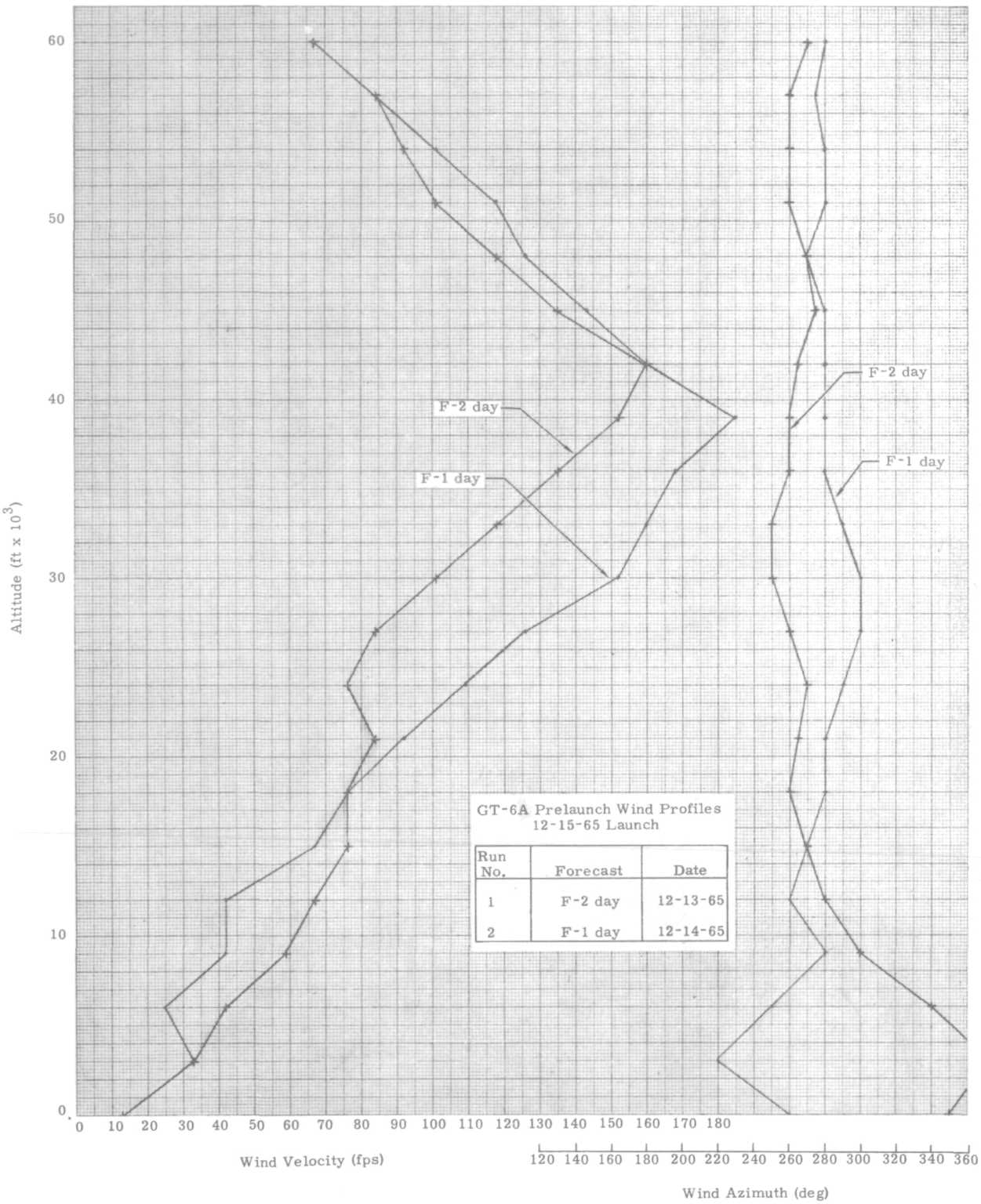


Fig. XI-5. F-2 and F-1 Day Wind Profiles

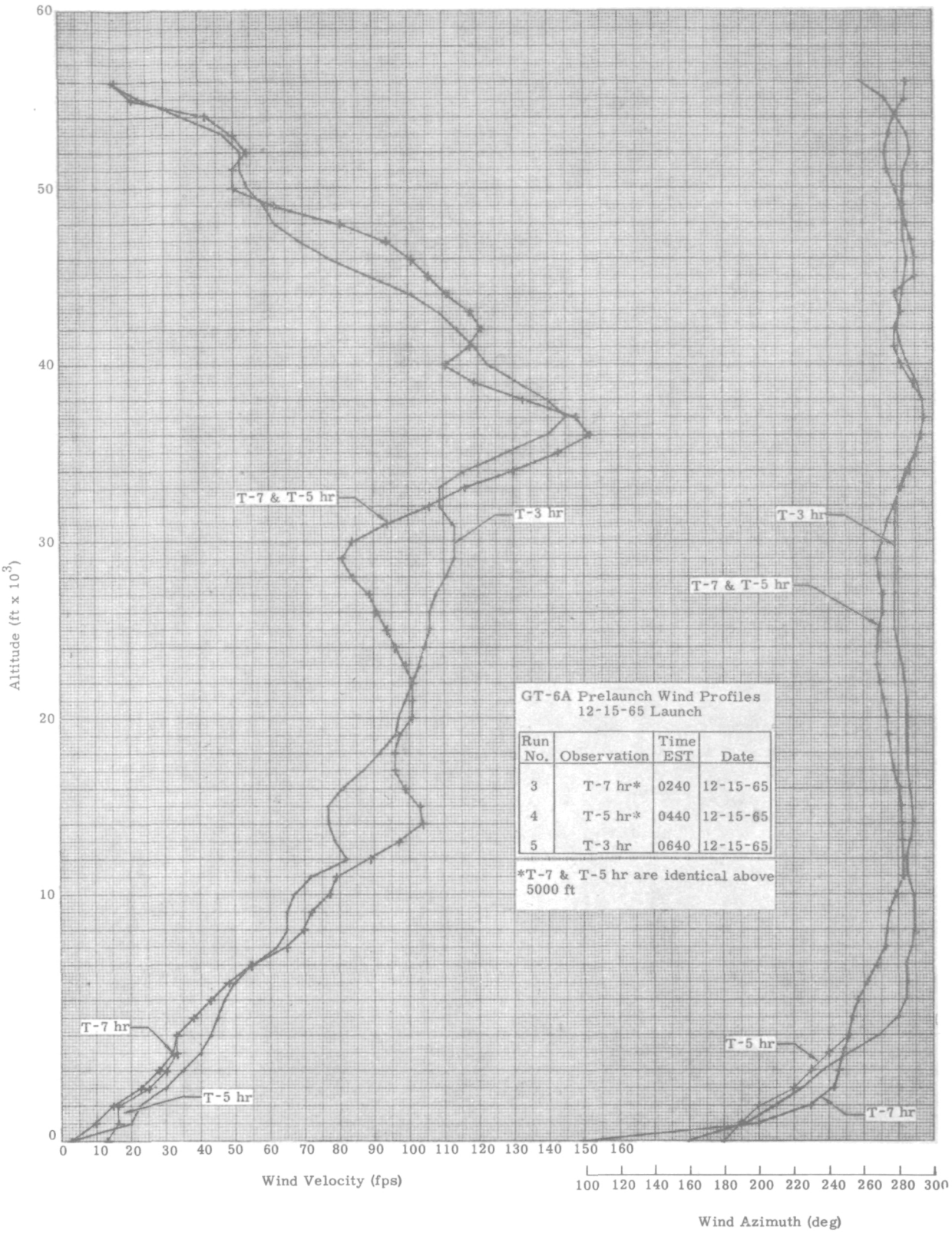


Fig. XI-6. T-7, T-5 and T-3 Hour Wind Profiles

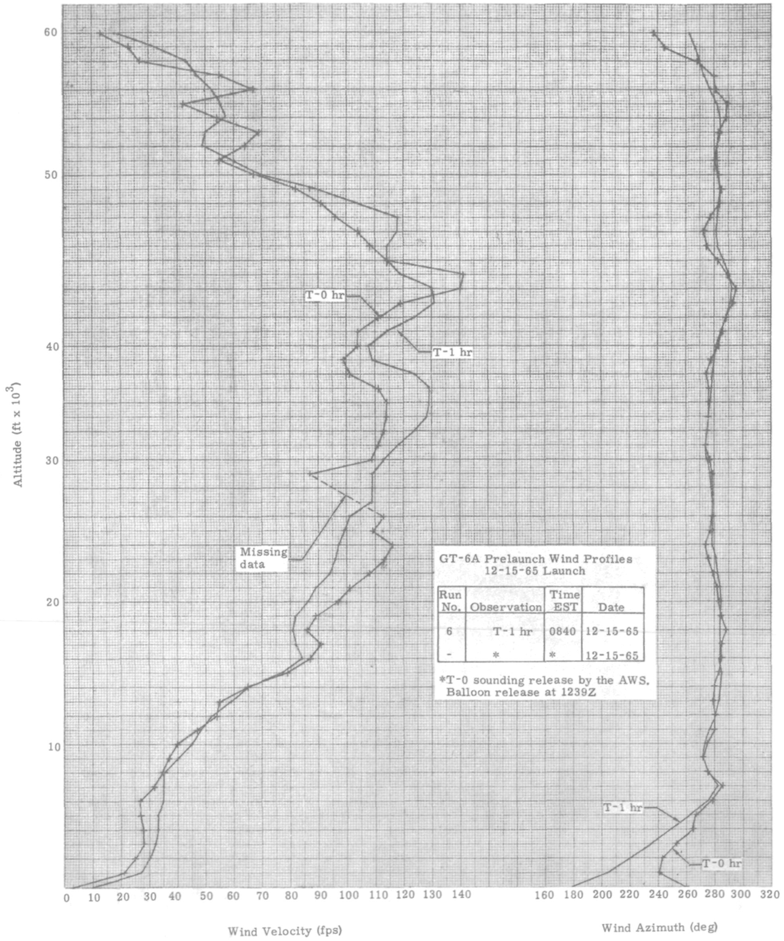
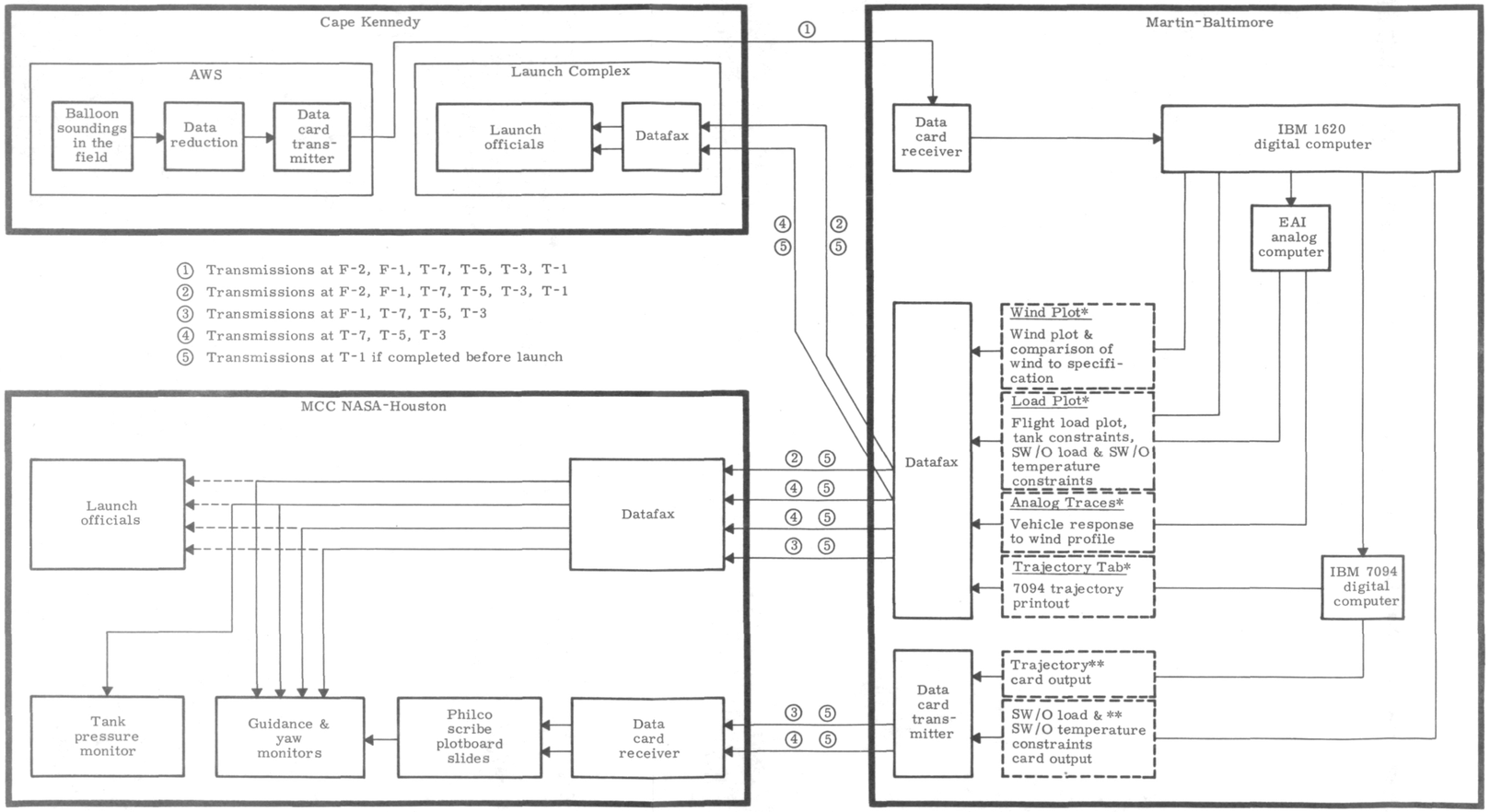


Fig. XI-7. T-1 and T-0 Hour Wind Profiles

XI-10-2



*Launch winds Datafax messages prepared by launch winds personnel

**Launch winds data card messages prepared by launch winds personnel

Fig. XI-8. Prelaunch Wind Operations

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TABLE XI-1 (continued)

Run No.	Time of Data Release to Martin-Baltimore	Operation
4	T-5 hours 0450 EST 12-12-65	Wind comparison to specification, trajectory, vehicle loads, vehicle response, tank underpressure constraints, switchover load constraints, and switchover temperature constraint were determined by computer programs. Data sent to Cape Kennedy and MCC-Houston. Wind in specification; load ratio 0.85; status go.
5	T-3 hours 0650 EST 12-12-65	Wind comparison to specification, trajectory, vehicle loads, vehicle response, tank underpressure constraints, switchover load constraint, and switchover temperature constraint were determined by computer programs. Data sent to Cape Kennedy and MCC-Houston. Wind in specification; load ratio 0.79; status go.
6	T-1 hour 0850 EST 12-12-65	Wind comparison to specification. Launch attempt occurred before the other simulations were completed. Wind in specification; status go. Status phoned to Cape Kennedy and MCC-Houston.

TABLE XI-2

Summary of Prelaunch Operations--GT-6A Launch
(15 December 1965)

Run No.	Time of Data Release to Martin-Baltimore	Operations
1	F-2 days 1100 EST 12-13-65	Wind comparison to specification. Sent to Cape Kennedy and MCC-Houston. Wind in specification; status go.
2	F-1 day 1100 EST 12-14-65	Wind comparison to specification and trajectory simulation. Sent to Cape Kennedy and MCC-Houston. Wind in specification; status go.

TABLE XI-2 (continued)

Run No.	Time of Data Release to Martin-Baltimore	Operations
3	T-7 hours 0240 EST 12-15-65	Wind comparison to specification, trajectory, vehicle loads, vehicle response, tank underpressure constraints, switchover load constraint and switchover temperature constraint were determined by computer programs. Data sent to Cape Kennedy and MCC-Houston. Wind in specification; load ratio 0.81; status go.
4	T-5 hours 0440 EST 12-15-65	Wind data identical to T-7 hour release except for ground winds. No computer runs. No-change status phoned to Cape Kennedy and MCC-Houston.
5	T-3 hours 0640 EST 12-15-65	Wind comparison to specification, trajectory, vehicle loads, vehicle response, tank underpressure constraints, switchover load constraint and switchover temperature constraints were determined by computer programs. Data sent to Cape Kennedy and MCC-Houston. Wind in specification; load ratio 0.80; status go.
6	T-1 hour 0840 EST 12-15-65	Wind comparison to specification. Launch occurred before other simulations were completed. Wind in specification; status go. Status phoned to Cape Kennedy and MCC-Houston.

C. SLOW MALFUNCTION MONITORING

1. Prelaunch Activities

From 6 December to 11 December, the guidance monitor activities included slide preparation, scriber data checks, plotboards for the staff service room and coordination with Philco and NASA.

2. Launch Day Activities (GT-6A Attempt)

All data card transmissions from Martin-Baltimore to NASA-Houston were received on schedule along with Datafax material. Four payload margin predictions were made by MRL and received from

Cape Kennedy verbally and by Datafax. Transmission No. 2 yielded the payload margin corresponding to the 600-pound Stage I pitch and 500-pound Stage II pitch and yaw payload constraint lines. The T-5 and T-3 hour structural temperature constraints were scribed on the 10 x 10 plotboard.

3. Launch Phase (GT-6A Attempt)

The liftoff signal was received at the Mission Control Center in Houston approximately 1.0 second after engine start; the subsequent shutdown signal followed immediately. Information based on the displays manned by the booster systems engineer and the tank monitor indicated that the thrust did not reach 35%. However, guidance monitor displays indicated that the programmers started in both the TARS and IGS.

Guidance monitor displays and communication were adequate for the GT-6A launch attempt.

4. Prelaunch Phase (GT-6A Launch)

Prelaunch activities started by scribing the T-3 hour trajectory on the 10 x 10 plotboard. All data card transmissions from Martin-Baltimore to NASA-Houston were received on schedule along with Datafax material; three payload margin transmissions and two sets of target data transmissions were received from Cape Kennedy. Plotboard switchover lines were identical to those determined for the 12 December launch attempt. At T-22 seconds, Meas 0153 indicated that actuator 4_1 started drifting from null in a command nose-down direction. The guidance monitor attempted to verify the validity of the telemetry system with Tel 3; however, a reply was not received prior to liftoff. At engine ignition the trace indicated normal operations.

5. Launch Phase (GT-6A Launch)

No indication of IGS gain change was present; however, if subsequent malfunctions had occurred, switchover would have been recommended. The Stage I lateral plotboard indicated a -200-ft/sec velocity at BECO, which probably resulted in the 0.074-degree wedge angle at insertion. Stage II pitch and yaw inboards were nominal. All analog displays and parameters were nominal except for the thrust attitude error in pitch, $\delta\beta_p$, which was not received from Burroughs, and a roll thrust misalignment of approximately 0.6-degree attitude error.

From the guidance monitor's viewpoint, GT-6A was the best flight to date. The V-A and III-A plotboards are shown in Figs. XI-9 and XI-10, respectively. Telemetry III yaw-roll and pitch axis recordings are shown in Figs. XI-11 and XI-12, respectively.

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XI-14-2

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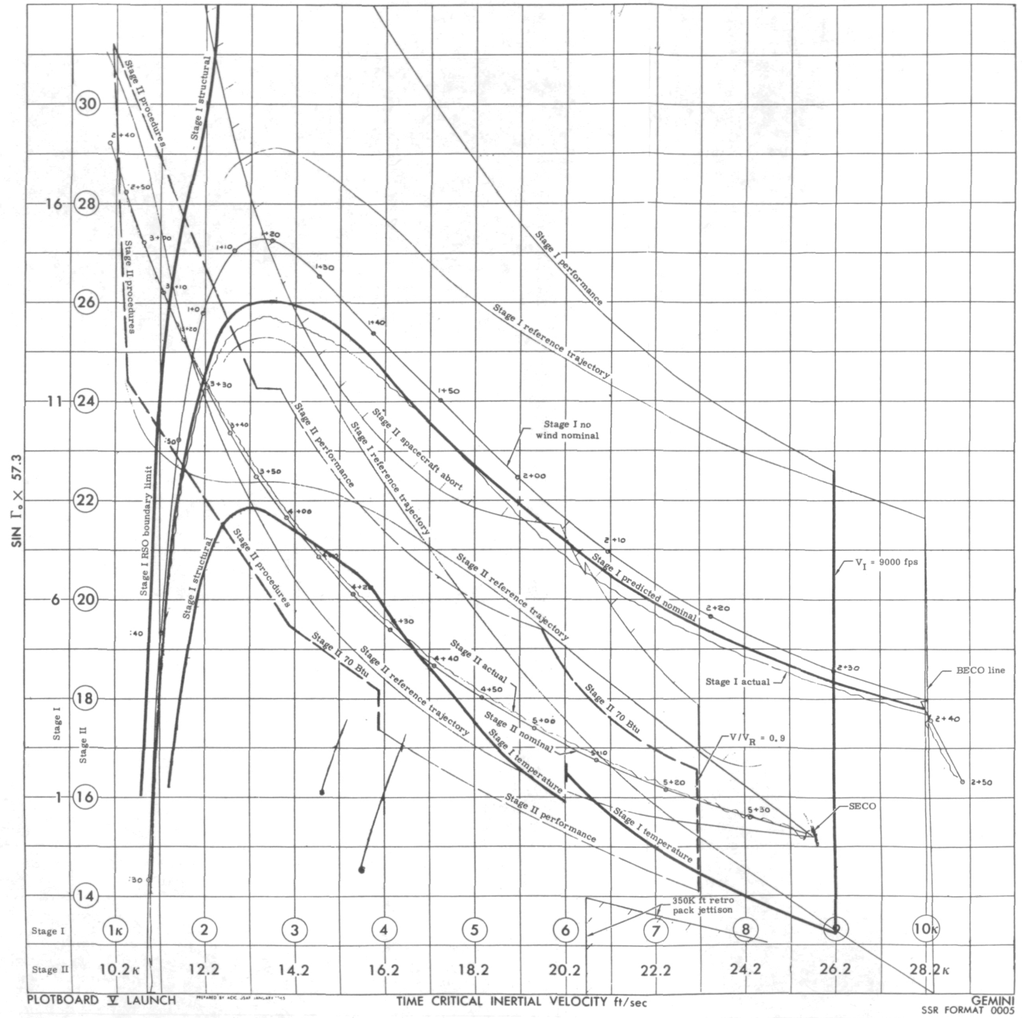

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Fig. XI-9. Houston-MCC Plotboard VA, Pitch Plane

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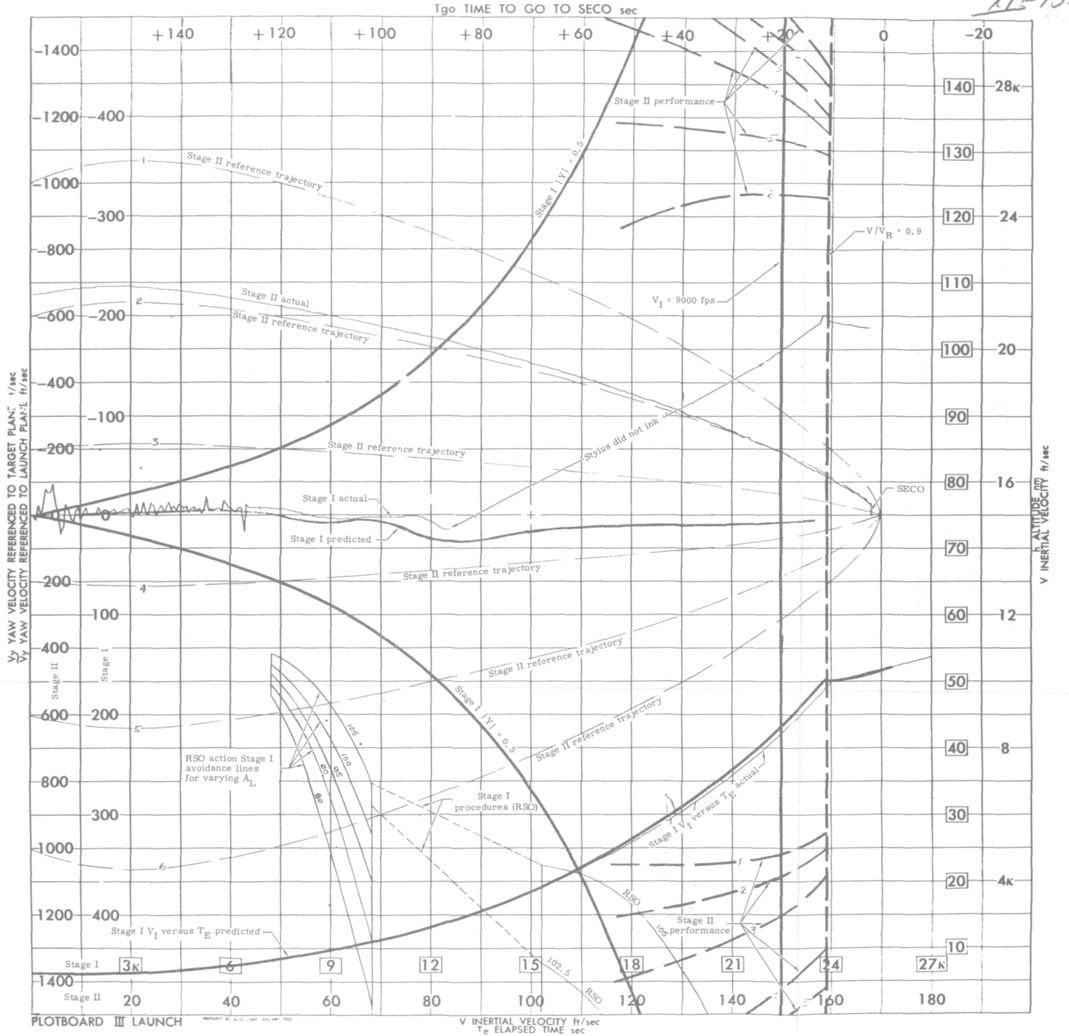

Fig. XI-10. Houston-MCC Plotboard III A, Lateral Velocity

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	Stage II	Stage I	
	1-Sec Marks from Liftoff		1
Roll CW ↓	ϕ_{e_s} 745	ϕ_{e_s} 745 ± 6°	1
—	SWO 872 or 874		2
	Diff 745 - 768	Diff 745 - 768	2
—	GCP 728		3
Yaw Right ↓	ψ_{e_s} 744	ψ_{e_s} 744 ± 6°	3
—	GCS 773		4
Yaw Roll Axis	Diff ψ_{e_s} - ψ_{e_p} 744 - 767	Diff ψ_{e_s} - ψ_{e_p} 744 - 767	4
—	Guid Init TARS 740		5
Yaw Right ↓	↑ 2°/sec W_{YN} GE/B	$\delta\psi_1$ 152 ± 5°	5
—	δ_6 GE/B		6
Yaw Right ↓	↑ 2°/sec $\dot{\psi}_g$ 756	APψ_1 769 MSB	6
—	δ_7 SSCO GE/B		7
Yaw Right ↓	$\delta\psi$ 650 ± 3°	$\delta\psi_2$ 151 ± 5°	7
—	SWO Command		8
Roll CW ↓	$\delta\phi$ 652 ± 33°	APψ_2 772 MSB	8

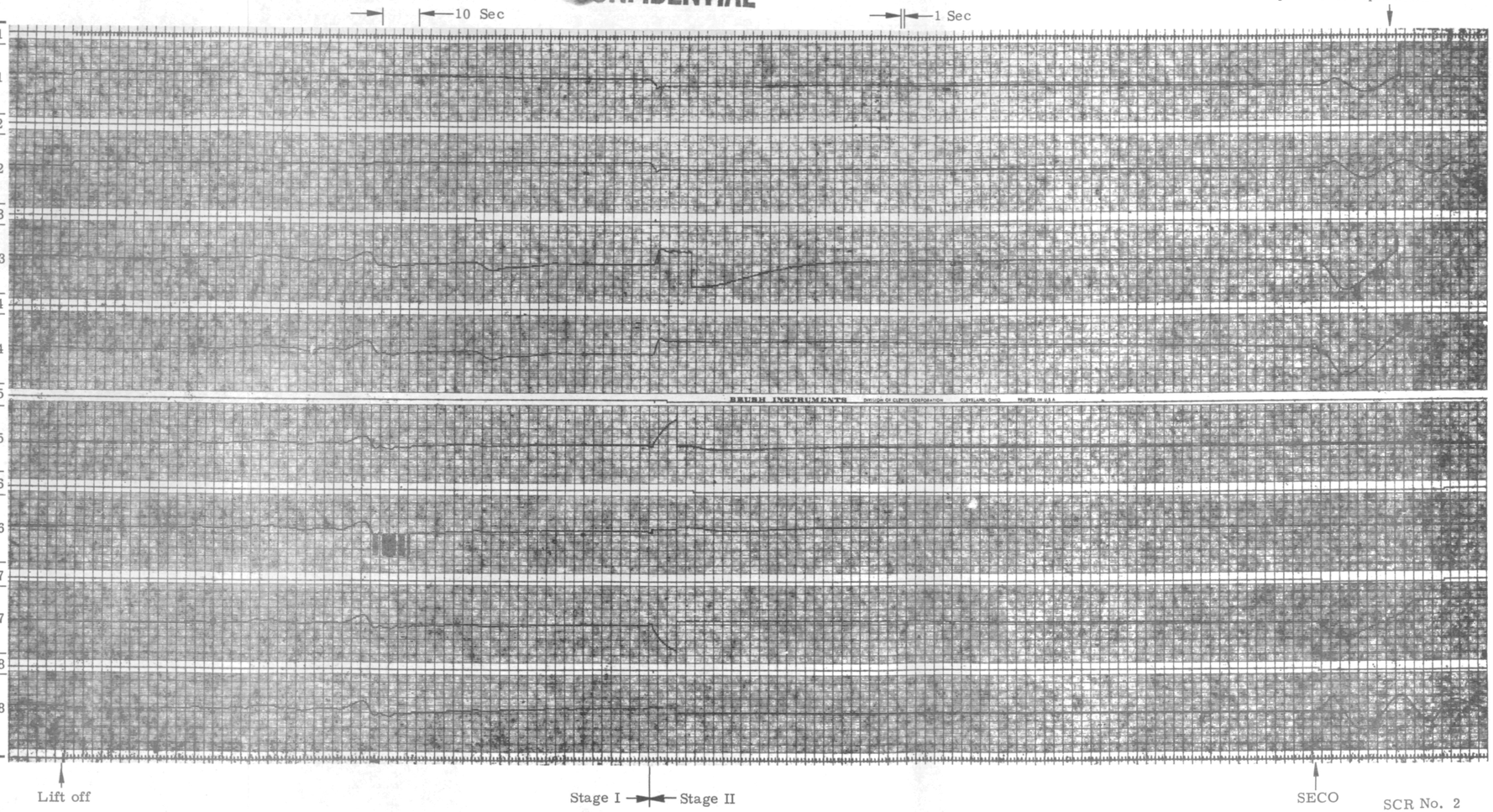

Fig. XI-11. Telemetry III Yaw-Roll Axis Recording

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	Stage II	Stage I	
	1-Sec Marks from Liftoff		1
Nose Down ↓	θ_{e_s} 743 ± 6°	θ_{e_s} 743 ± 6°	1
	SWO 872 - 874		2
—	Diff $\theta_{e_s} - \theta_{e_p}$ 743 - 766	Diff $\theta_{e_s} - \theta_{e_p}$ 743 - 766	2
	GCP 728		3
Stage II ↓ Nose Down ↑ Stage I	δB_p GE/B ± 5°	$\dot{\theta}_c$ 732 MSB	3
	GCS 773		4
Nose Down ↓	ω_{pN} GE/B ±2°/sec	$\delta\theta_1$ 153 ± 5°	4
	Guid Init TARS 740		5
Nose Down ↓	$\dot{\theta}_g$ 755 ±2°/sec	$AP\theta_1$ 770 MSB	5
	δ_6 Guid Start GE/B		6
Stage I ↓ Nose Down ↑ Stage II	ω_{ps} GE/B ±1°/sec	$\delta\theta_2$ 150 ± 5°	6
	δ_7 SSCO GE/B		7
Nose Down ↓	$\delta\theta$ 651 ± 3°	$AP\dot{\theta}_2$ 771 MSB	7
	SWO Command		8
Roll CW ↓	———	$\dot{\phi}_c$ 734 MSB	8

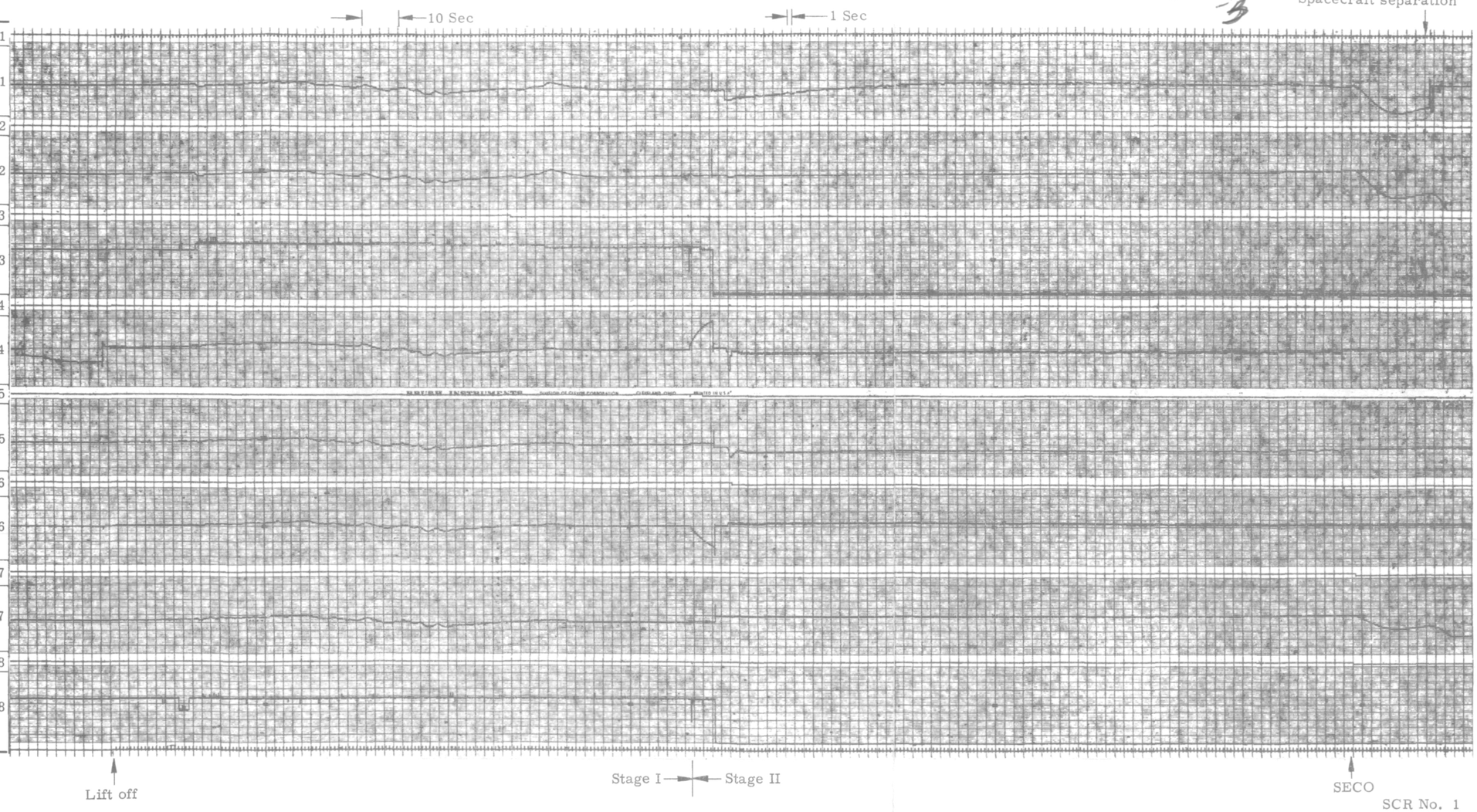

Fig. XI-12. Telemetry III Pitch Axis Recording

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XII. AIRFRAME SYSTEM

A. STRUCTURAL LOADS

Analysis of GT-6A flight data indicates that the loads experienced were well within the structural capability of the launch vehicle. The most critical loading occurred, characteristically, at pre-BECO where the load aft of Station 320 reached 103.6% of design limit load in compression (DLL_c). Instrumentation for dynamic response data consisted of rate gyros for lateral dynamic loads and axially mounted accelerometers for longitudinal dynamic loads. No major anomalies affecting the airframe occurred during flight; unusually high amplitude third structural mode lateral oscillations occurred just prior to BECO but were not considered to be detrimental.

1. Preignition

The 1 g deadweight distribution is the only contribution to steady axial loading in the preignition period. Ground winds were approximately 7 mph from a direction of 210 degrees, resulting in steady bending of 330,000 in. -lb and wind-induced oscillatory (WIO) loads of ±40,000 in. -lb (Fig. XII-1) at Station 1224. The WIO response represents approximately 2% of the WIO design limit bending moment; Table XII-1 shows the comparison of GLV WIO experience to date.

TABLE XII-1

Comparison of GLV WIO Loads

Flight	WIO Load at Station 1224 (% of WIO design limit bending moment)
GT-1	52
GT-2	5
GT-3	29
GT-4	3
GT-5	2
GT-7	40
GT-6A	2

2. Launch Prerelease

Ignition transients were normal, and the attendant dynamic axial loads as measured by the BLH system are shown in Fig. XII-2 together with the steady axial load. The prerelease lateral dynamic loading was

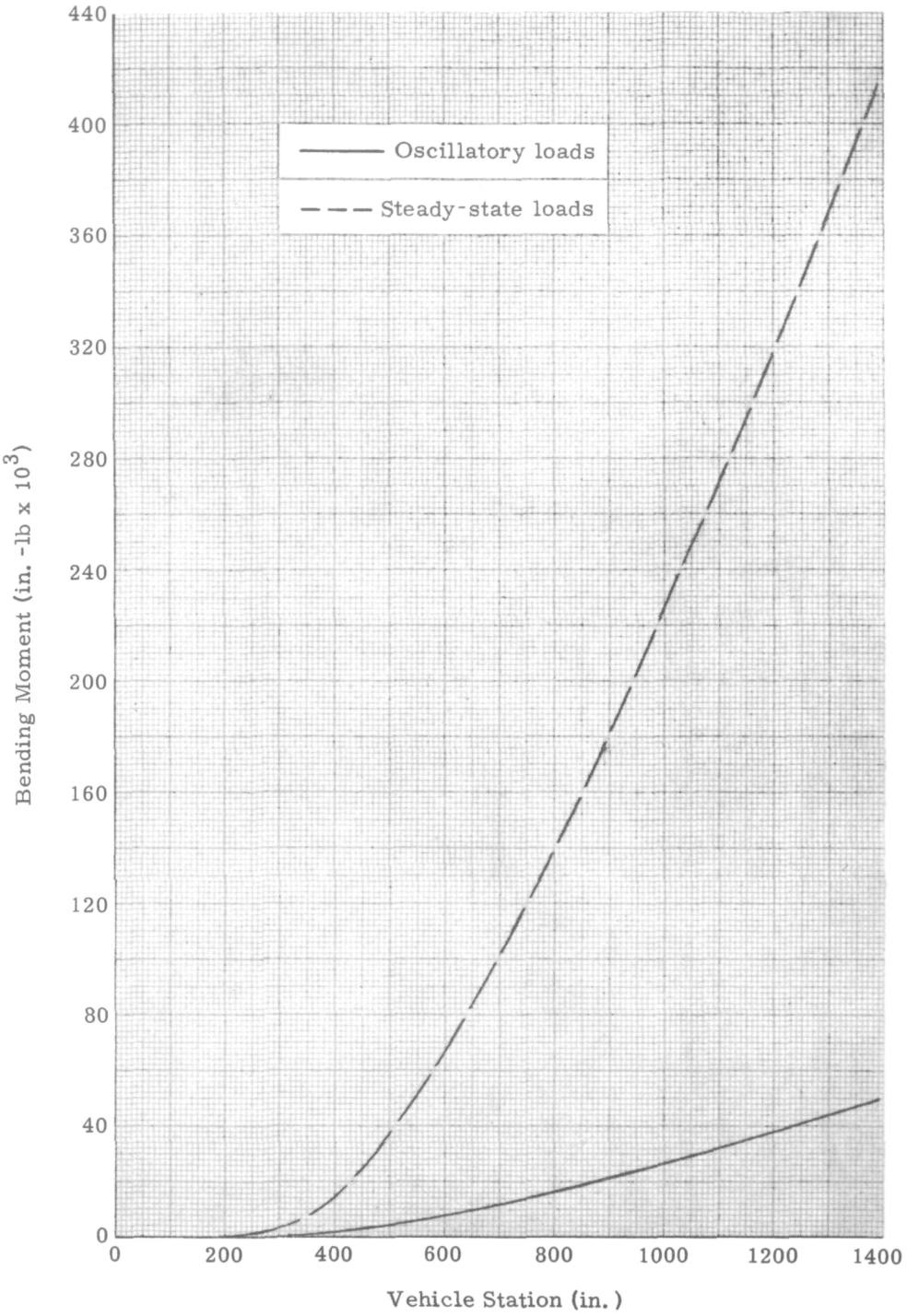


Fig. XII-1. Bending Moments Due to Ground Winds: Preignition

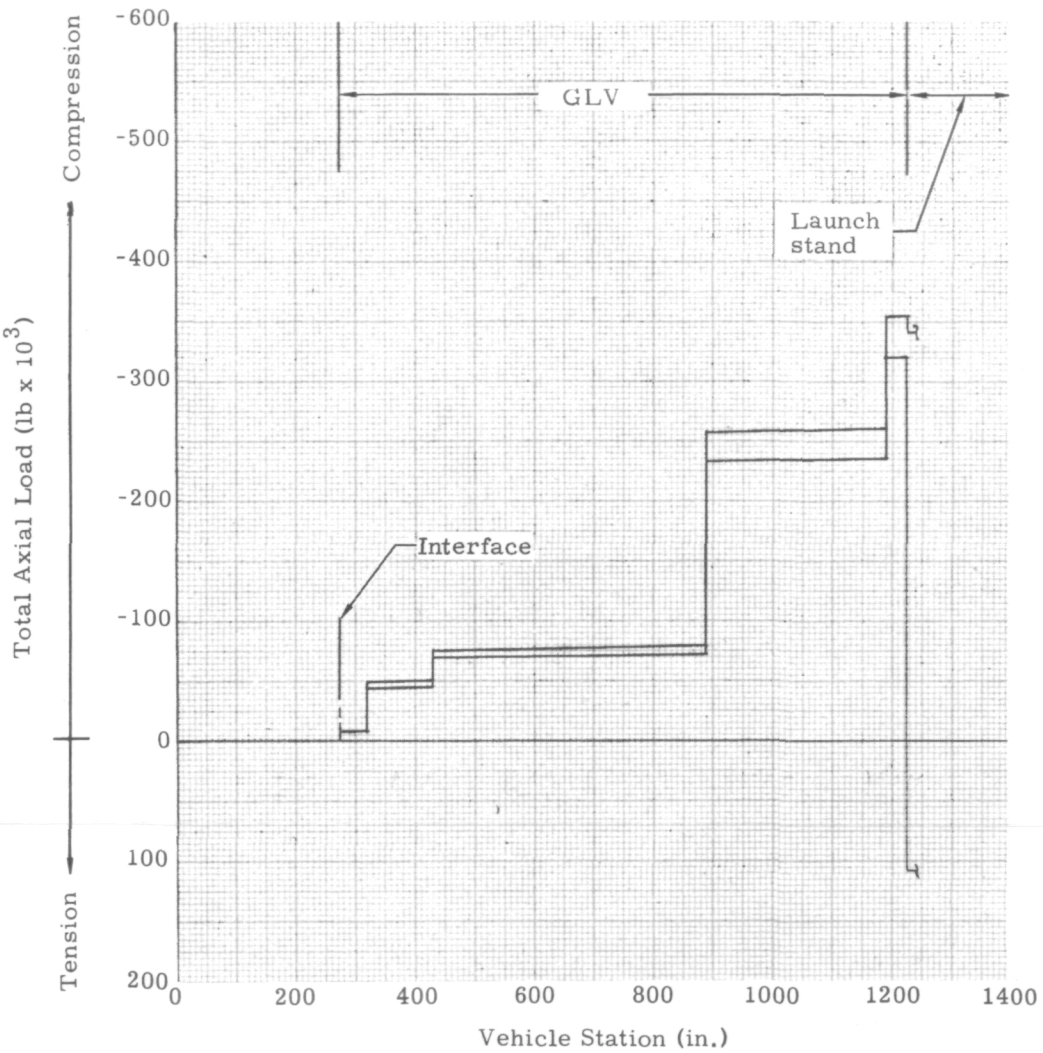


Fig. XII-2. Dynamic Axial Load Envelope: Prelaunch

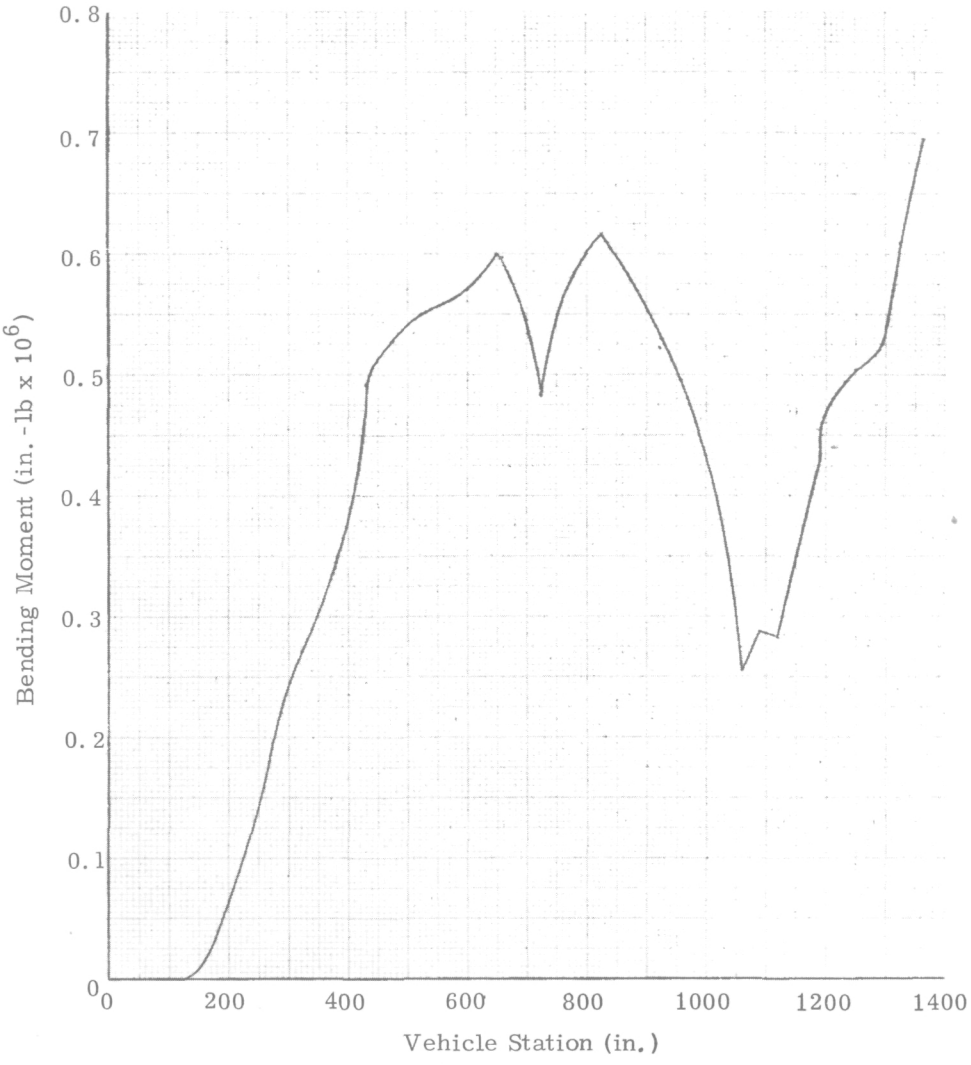


Fig. XII-3. Lateral Dynamic Bending Moment Envelope: Prerelease

due to the combined effects of ground winds and engine start transients; this loading is shown in Fig. XII-3.

3. Launch Postrelease

A comparison (Table XII-2) of the GT-6A liftoff load factor with those of previous launches indicates that this flight experienced the lowest initial steady acceleration to date. This can be attributed to the low thrust class Stage I engine.

TABLE XII-2

Comparison of GLV Liftoff Load Factors

Flight	Liftoff Load Factor (g)
GT-1	1.27
GT-2	1.27
GT-3	1.27
GT-4	1.27
GT-5	1.28
GT-7	1.26
GT-6A	1.25

Dynamic deformation modes in evidence at postrelease consisted of the first and third structural bending and Stage I engine modes in the lateral plane and the first axial mode in the longitudinal direction. Frequency correlation between calculated and observed modes during the flight is given in Fig. XII-4; the resulting dynamic bending moment in the postrelease condition is shown in Fig. XII-5.

4. Stage I Flight

The most significant periods of Stage I flight for airframe loading occurred at Max C_{N_α} $q\alpha$ and at pre-BECO. Max C_{N_α} $q\alpha$ occurred at LO + 80 seconds, the same time in flight as on GT-7. A 35-fps wind shear spike at an altitude of 43,000 feet accounted for this late occurrence.

In comparing loads at LO + 80 seconds with loads at the traditional LO + 69 seconds Max C_{N_α} $q\alpha$ flight time, several interesting observations can be made:

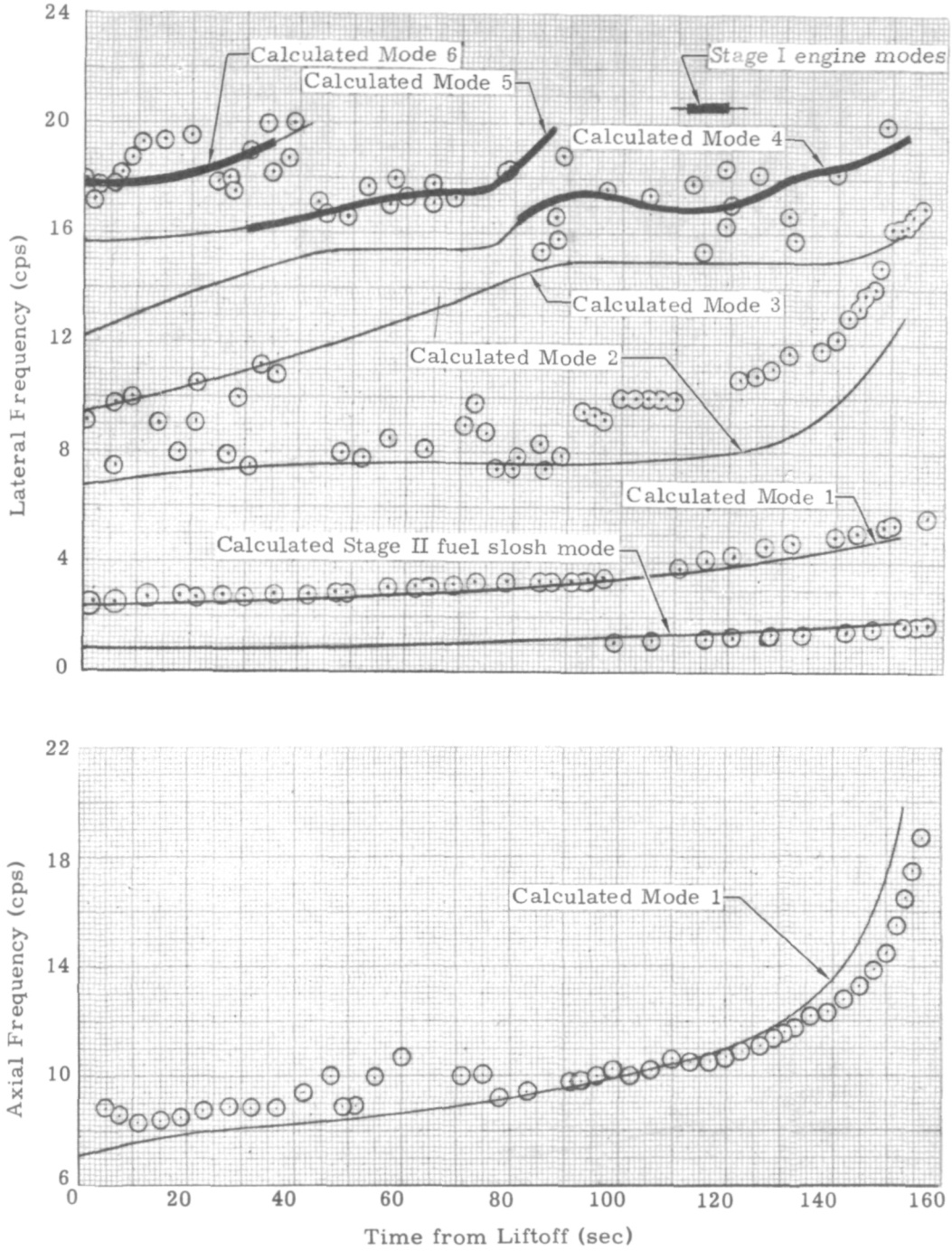


Fig. XII-4. Stage I Flight Vibration Frequency Correlation

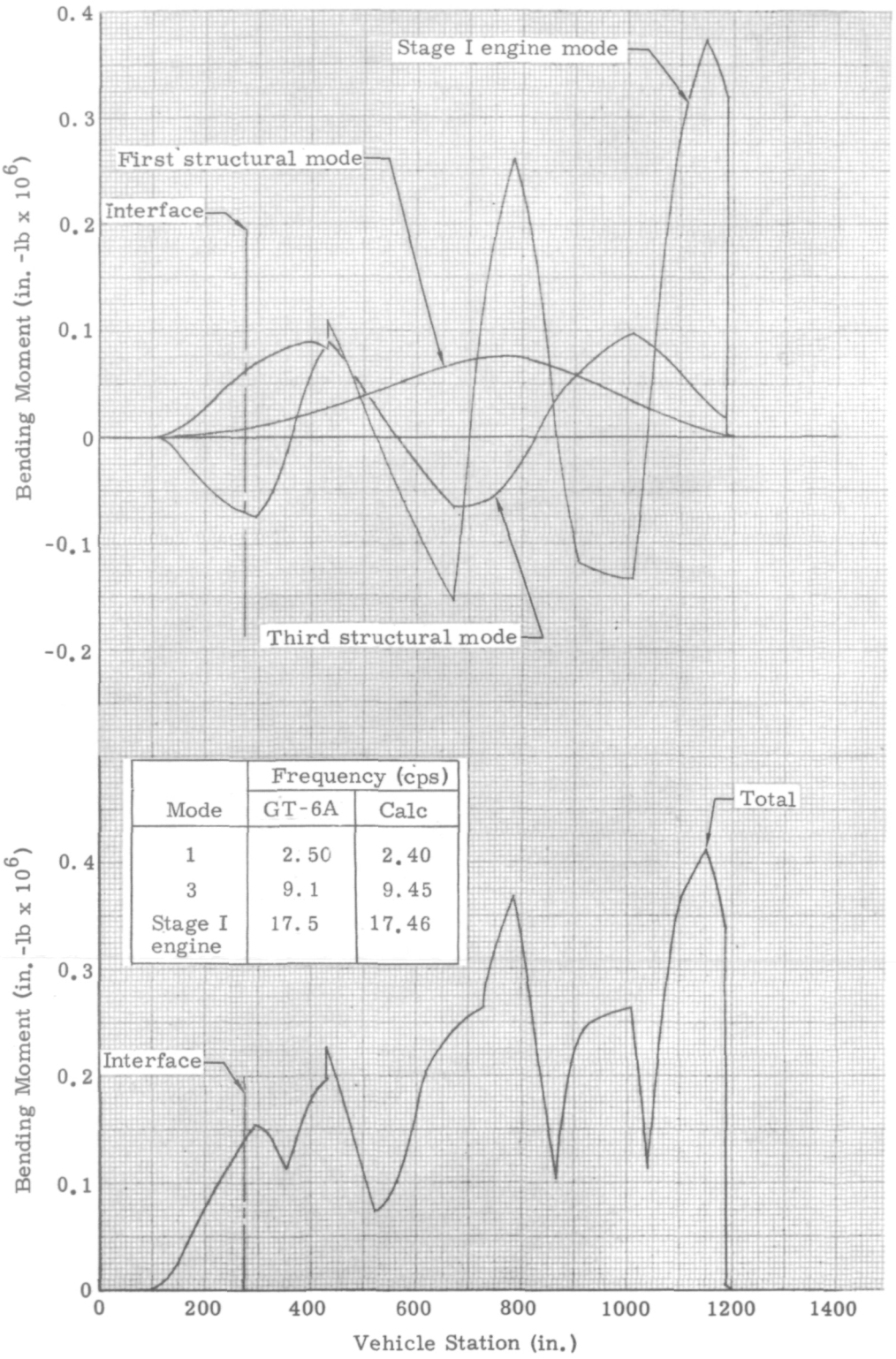


Fig. XII-5. Total Lateral Dynamic Load Envelope: Postrelease

(1) All of the lateral dynamic responses noted at LO + 69 seconds were also observed at LO + 80 seconds. The amplitudes of the individual vibration modes at LO + 69 seconds were significantly greater than for corresponding modes at LO + 80 seconds to the extent that the total lateral dynamic load increment at LO + 69 seconds was three times that at LO + 80 seconds.

(2) The steady axial acceleration at LO + 80 seconds was greater than at LO + 69 seconds. Because of the low thrust of the GT-6A Stage I engine and the additional depletion of propellants in that 11-second period, the resultant quasi-steady axial load at LO + 80 seconds was somewhat less than on previous flights at LO + 69 seconds.

(3) Winds aloft were the most severe since the flight winds experienced on GT-3. The wind magnitude at LO + 80 seconds was 2/3 of the design specification level, a wind shear spike was superimposed on the steady wind profile, and the wind azimuth was closer to the critical azimuth than on any flight subsequent to GT-3. Since the bending load peak moves forward on the launch vehicle as the vehicle cg moves forward with depletion of propellants, the quasi-steady bending loads at critical Station 935 were partially reduced from the extreme loading induced by the effects of the magnitude, shear spike, and azimuth of the winds.

The net result of the foregoing considerations was the attainment of a lower than average airframe loading at Max C_{N_α} $q\alpha$. This comparison is shown in Table XII-3.

TABLE XII-3

Structural Loads Comparison of Gemini Flights

Flight	At Max C_{N_α} $q\alpha$, Station 935 (% of DLL_c)	At Pre-BECO Station 320+ (% of DLL_c)
GT-1	82.3	95.5
GT-2	80.1	100
GT-3	78.5	97
GT-4	85.4	101
GT-5	71.6	98.5
GT-7	72.8	98.5
GT-6A	74.8	103.6

Dynamic bending moments obtained from the rate gyro responses in the Max $C_{N_\alpha} q\alpha$ and pre-BECO regions of flight are shown in Figs. XII-6 and XII-7, respectively. Steady axial acceleration at pre-BECO is given in Table XII-4.

Lateral oscillations associated with the third structural mode of the launch vehicle and occurring three seconds prior to BECO were approximately three times greater than those experienced on any previous flight. The cause of this unusually high level of oscillation, although at present unexplained, may be due to modal cross-coupling of the 16.7 cps lateral vibration and the inherent longitudinal oscillations (at 16.8 cps) during this time of flight. An analysis of the oscillations shows that the resultant dynamic load was 7% of the vehicle design limit load and is not of great concern since ample load margin remains.

TABLE XII-4

Steady Axial Accelerations at Pre-BECO

Flight	Pre-BECO Axial Acceleration (g)
GT-1	5.61
GT-2	5.69
GT-3	5.63
GT-4	5.63
GT-5	5.55
GT-7	5.56
GT-6A	5.46

5. Stage II Flight

Propellant slosh oscillations, associated with the calculated Stage II fuel slosh mode, occurred continuously from LO + 230 seconds to LO + 320 seconds; the amplitude of the oscillations was small (less than ±0.1 deg/sec), as indicated by both pitch and yaw rate gyros. No recurrence of the large amplitude (±0.6 deg/sec) GT-7 slosh oscillations was noted; the GT-6A oscillations were consistent with those on previous flights other than GT-7.

The steady axial acceleration at SECO is shown in Table XII-5 and was one of the lowest of any GLV flight. This can be attributed to a lower than predicted Stage II thrust level, to one of the lowest thrust levels at SECO on flights to date and to trajectory differences.

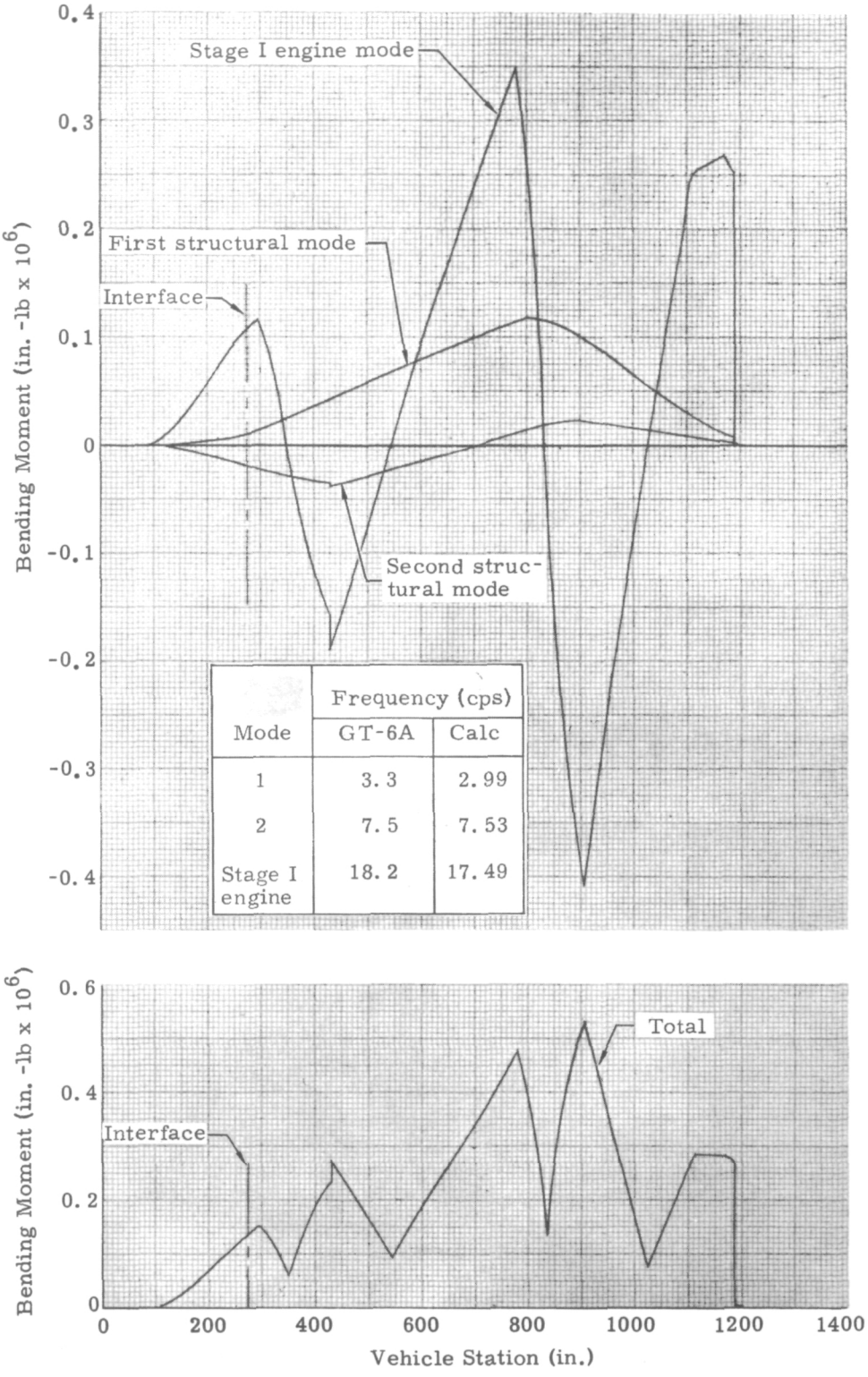
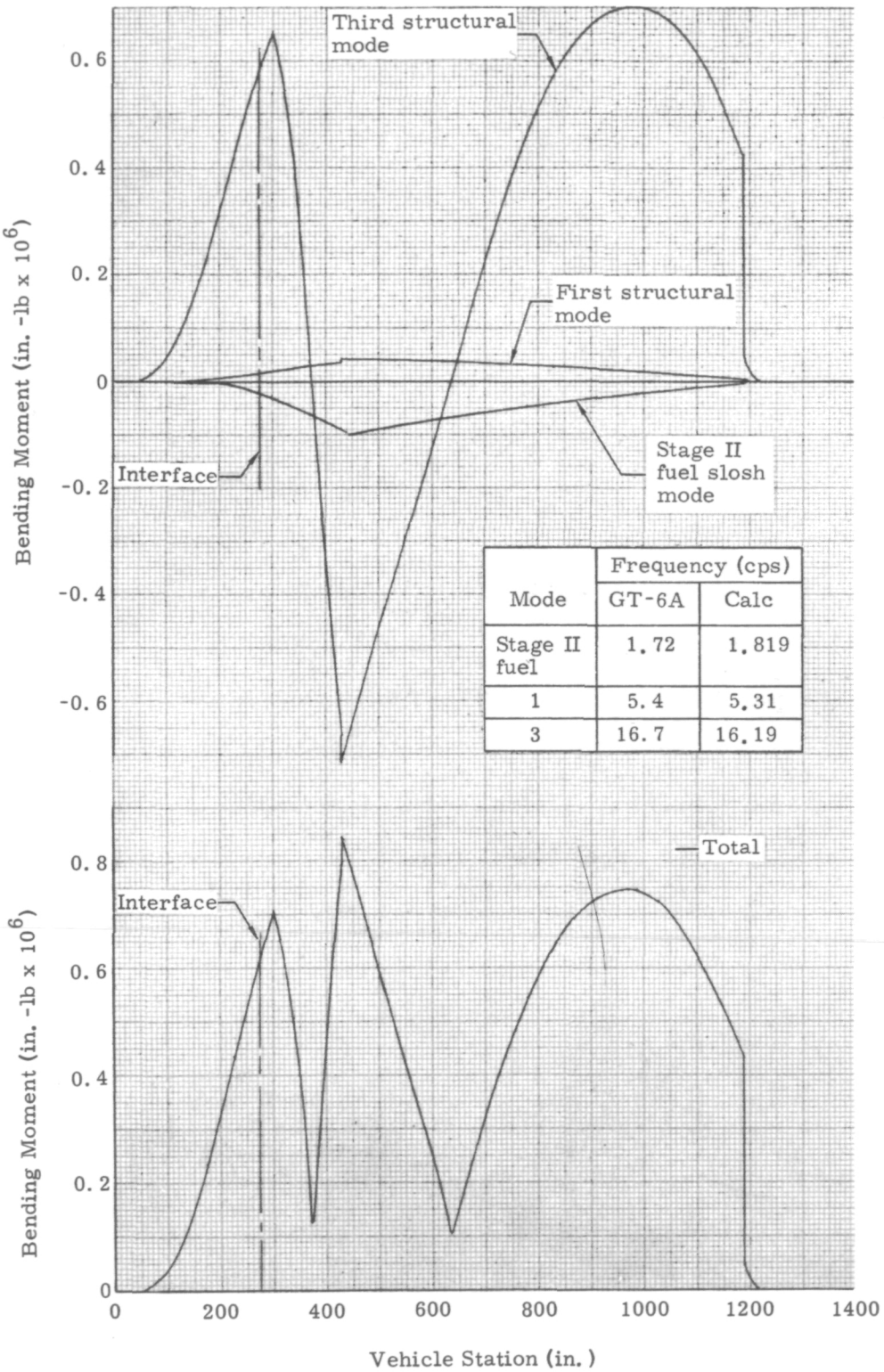


Fig. XII-6. Total Lateral Dynamic Load Envelope: Max C_{N_α} $q\alpha$



Mode	Frequency (cps)	
	GT-6A	Calc
Stage II fuel	1.72	1.819
1	5.4	5.31
3	16.7	16.19

Fig. XII-7. Total Lateral Dynamic Load Envelope: Pre-BECO

TABLE XII-5

Steady Axial Accelerations at SECO

Flight	SECO Axial Acceleration (g)
GT-1	7.35
GT-2	7.70
GT-3	7.50
GT-4	7.42
GT-5	7.56
GT-7	7.23
GT-6A	7.33

There were four indications on the low range (±0.5 g, Meas 0699) axial accelerometer of post-SECO disturbance. The times of occurrence of these disturbances and their associated levels are:

(1) Less than 0.02 g at SECO + 5.1 seconds

(2) Less than 0.02 g at SECO + 6.6 seconds

(3) Less than 0.1 g at SECO + 17.5 seconds

(4) Less than 0.1 g at SECO + 28.0 seconds.

The first two occurrences were noted only on the low range accelerometer, the third occurrence was noted on the low range accelerometer and one of the Stage II actuator measurements, and the last occurrence was noted on the low range accelerometer and on all actuators and rate gyros. Spacecraft separation from the sustainer occurred at SECO + 22.2 seconds.

6. Total Airframe Loads

A summary of the total airframe loads (quasi-steady axial, dynamic axial and equivalent axial loads from quasi-steady and dynamic bending moments) for significant structural loading conditions at critical stations is presented in Table XII-6. Complete vehicle loading at significant flight times is shown in Figs. XII-8 through XII-12. The maximum load at any station and the loading condition for which it occurred are shown in Fig. XII-13.

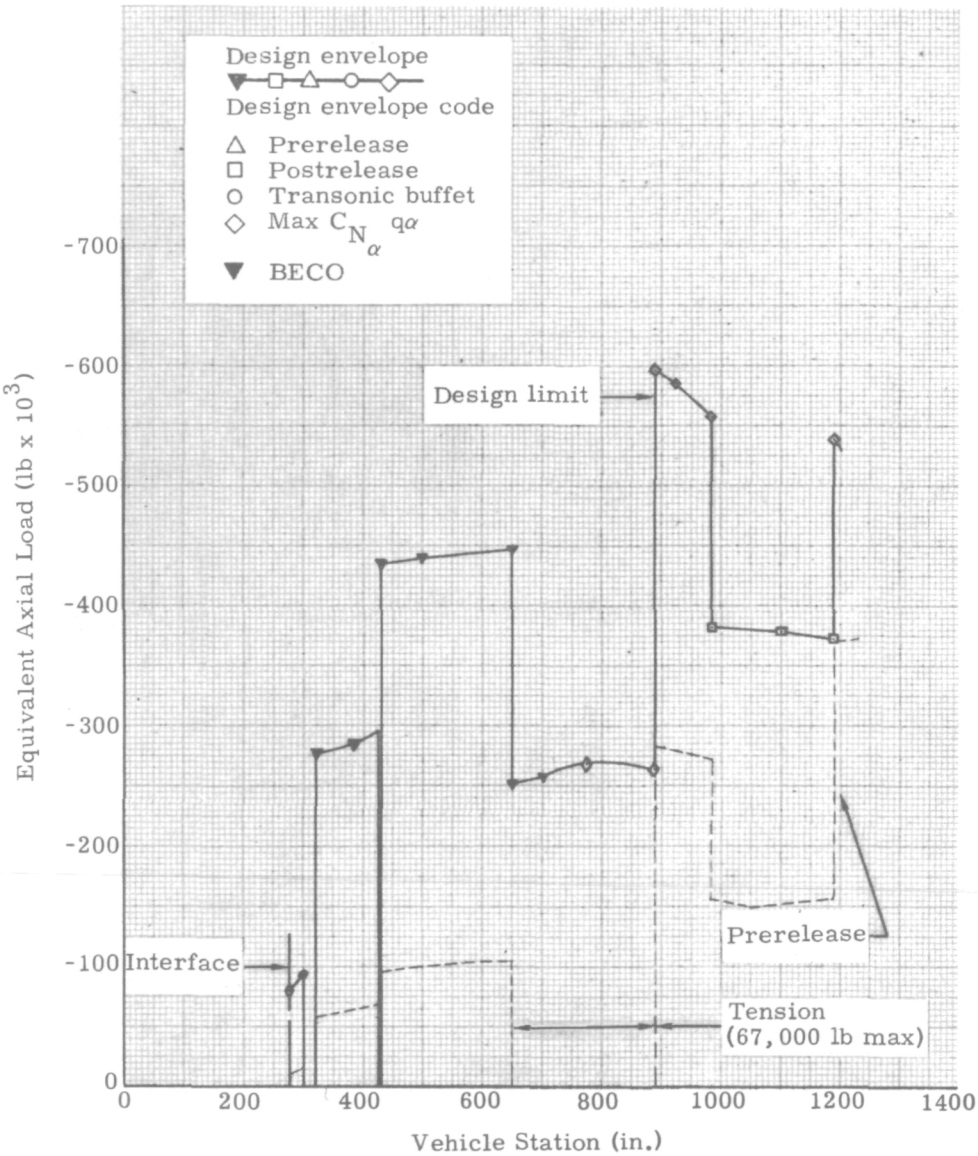


Fig. XII-8. Equivalent Axial Load: Prerelease

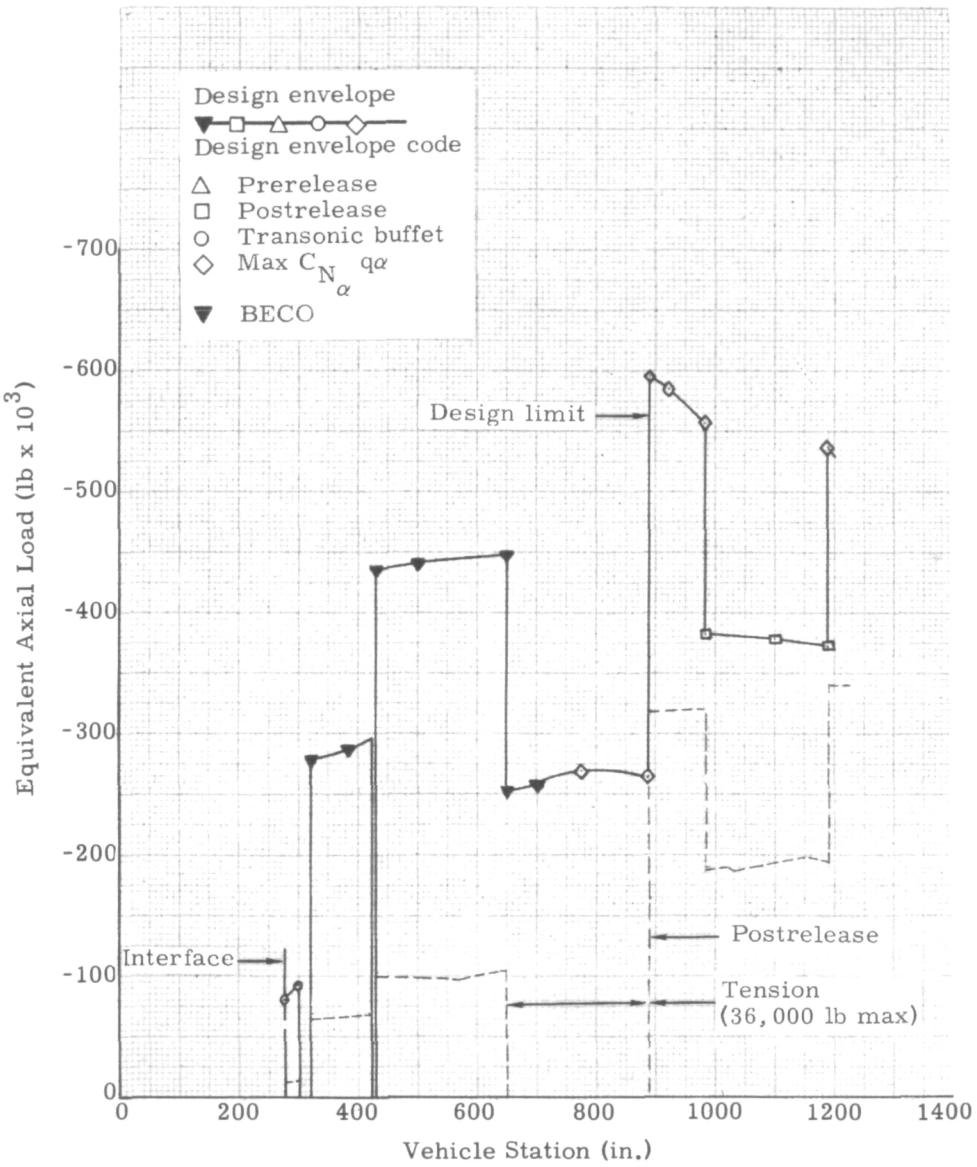


Fig. XII-9. Equivalent Axial Load: Postrelease

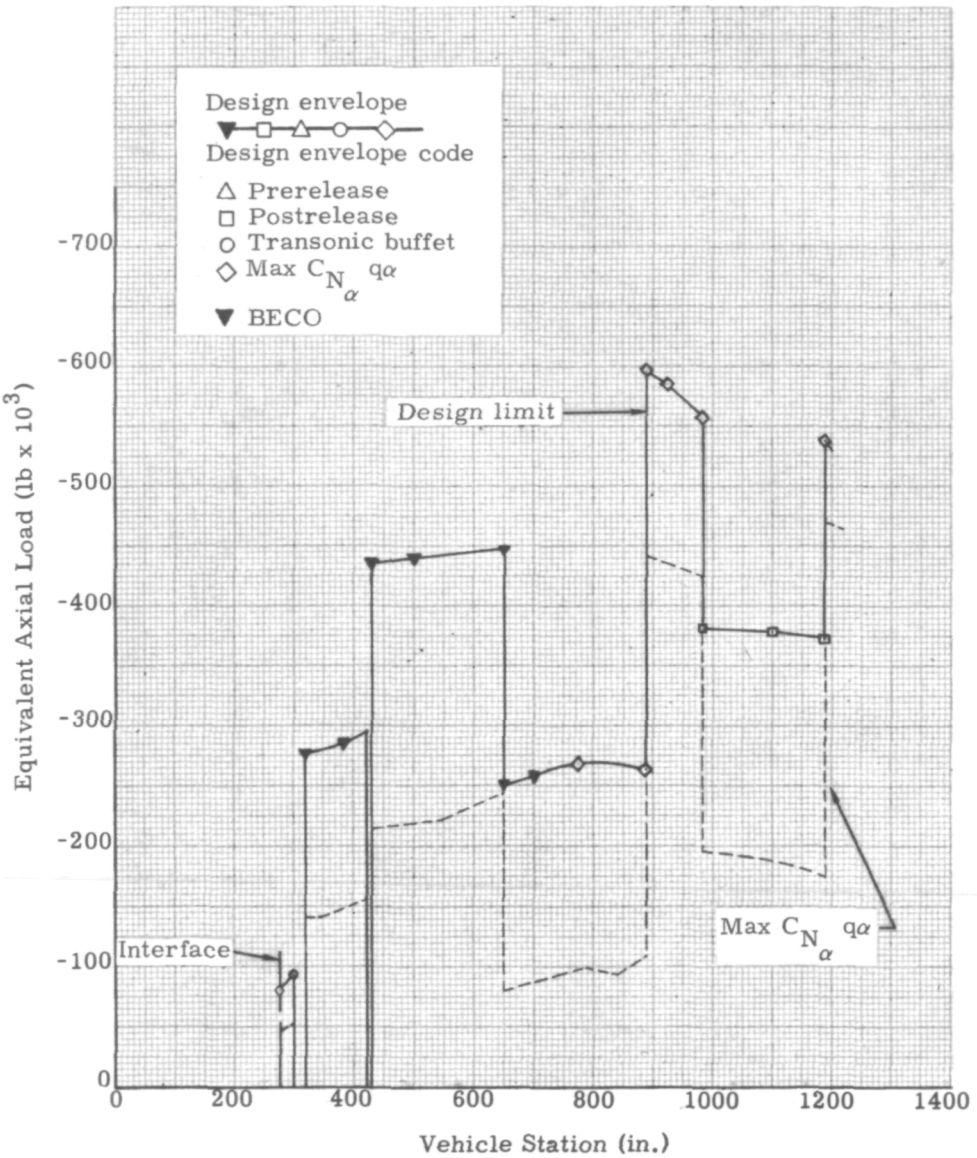


Fig. XII-10. Equivalent Axial Load: Max C_{N_α} $q\alpha$

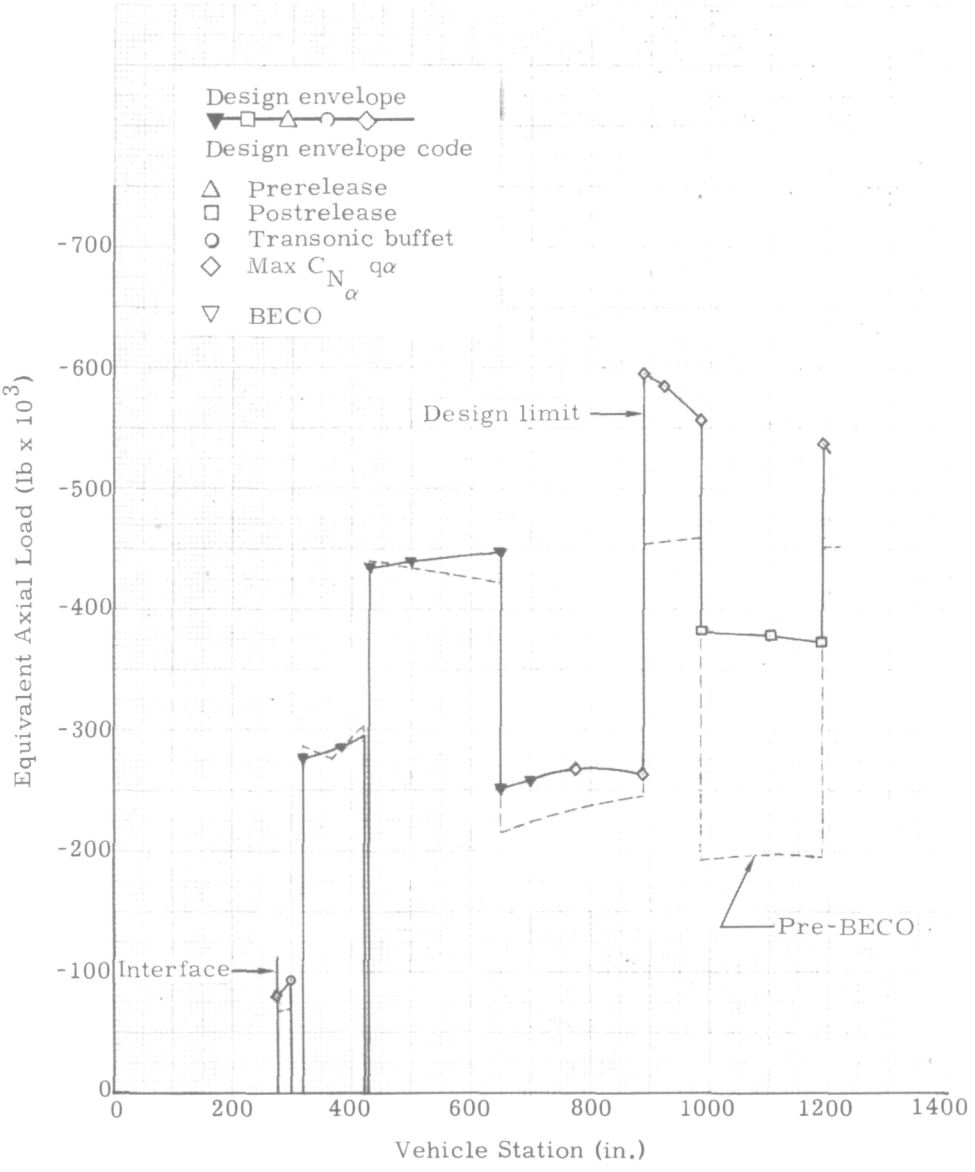


Fig. XII-11. Equivalent Axial Load: Pre-BECO

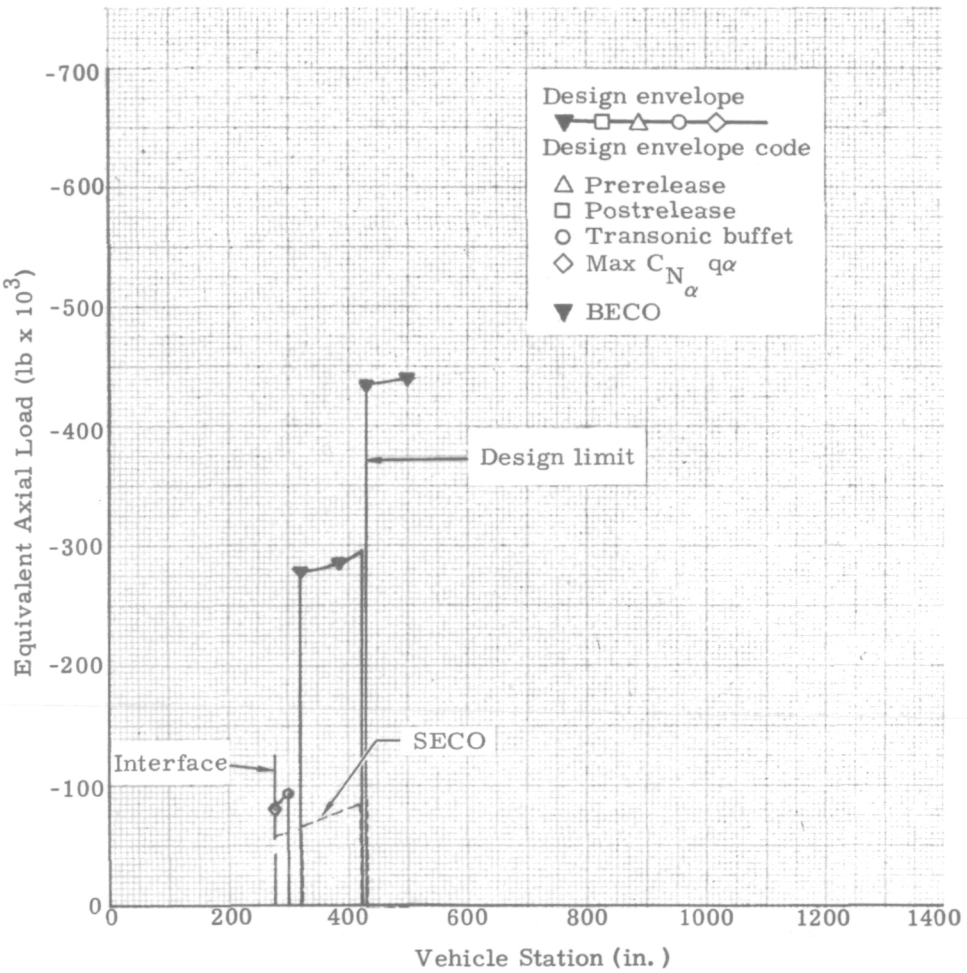


Fig. XII-12. Equivalent Axial Load: SECO

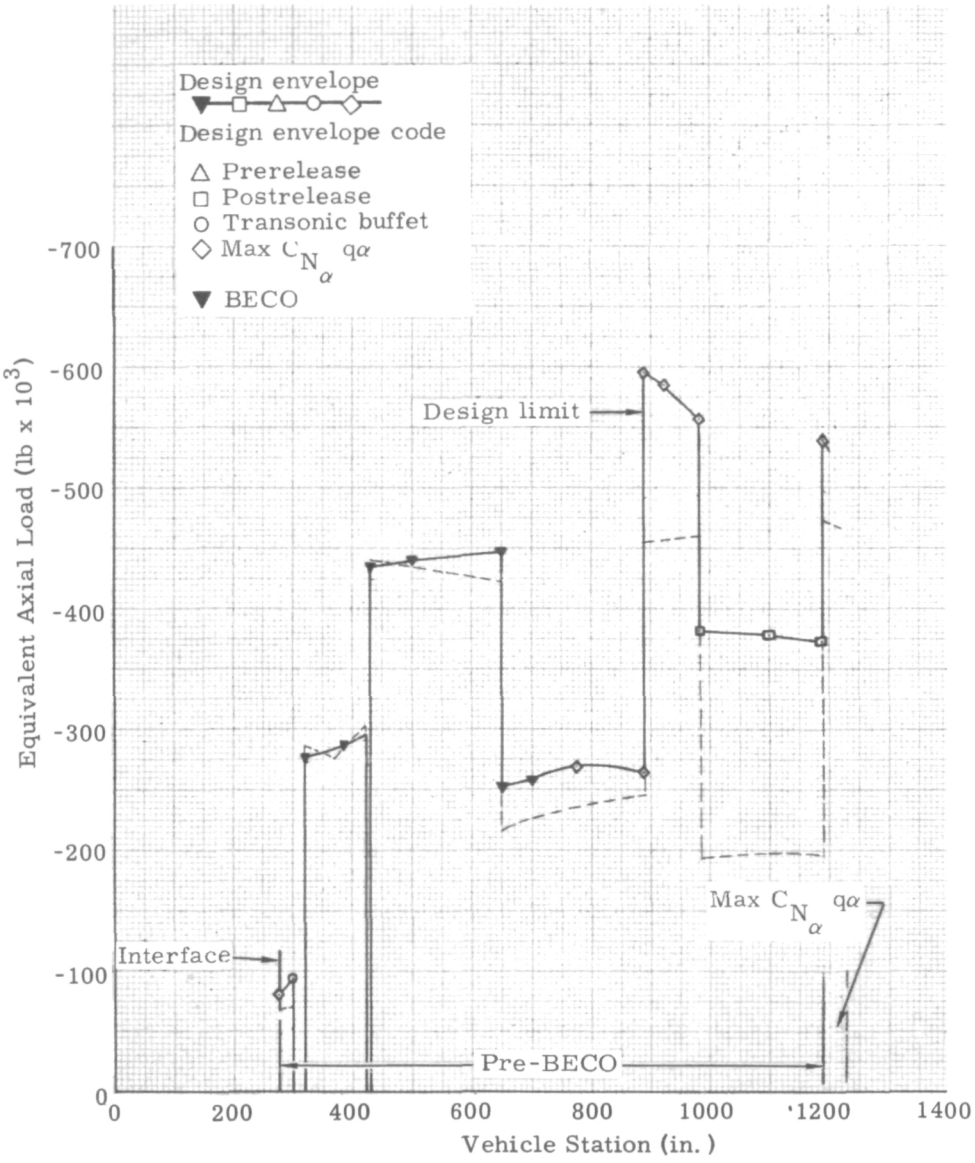


Fig. XII-13. Maximum Structural Load Envelope

TABLE XII-6

Summary of GT-6A Total Airframe Loads

Flight Condition	Total Airframe Load (% of DLL_c at critical station)	Critical Station
Prerelease	70.6	1188*
Postrelease	62.7	1188*
Max C_{N_α} $q\alpha$ (LO + 80 sec)	74.8 89.0	935 1188*
Pre-BECO (LO + 156 sec)	103.6	320*
Pre-SECO (LO + 338 sec)	71.7	276.8*

*Just aft of station.

B. POGO

Analysis of GT-6A telemetry data showed that the POGO suppression devices (fuel accumulators and oxidizer standpipes) operated satisfactorily. No pressure oscillations which could be associated with airframe structural resonances were detected in either the oxidizer or the fuel feedline.

Bandpass filtering of the analog reconstruction of PCM/FM telemetry (Meas 0670) indicates that the maximum intermittent longitudinal oscillation at the spacecraft-launch vehicle interface occurred at LO + 146.8 seconds and again at LO + 153.9 seconds. The amplitude of these oscillations was 0.115 g zero-to-peak, with corresponding response frequencies of 13.7 cps and 16.8 cps, respectively, and lasted approximately one to two seconds in each instance. The time history of POGO response amplitude for Compartments 1 and 5 is shown in Fig. XII-14.

Figure XII-15 is a comparison of longitudinal oscillations for the last four Gemini flights. The GT-6A level represents the lowest POGO occurrence to date.

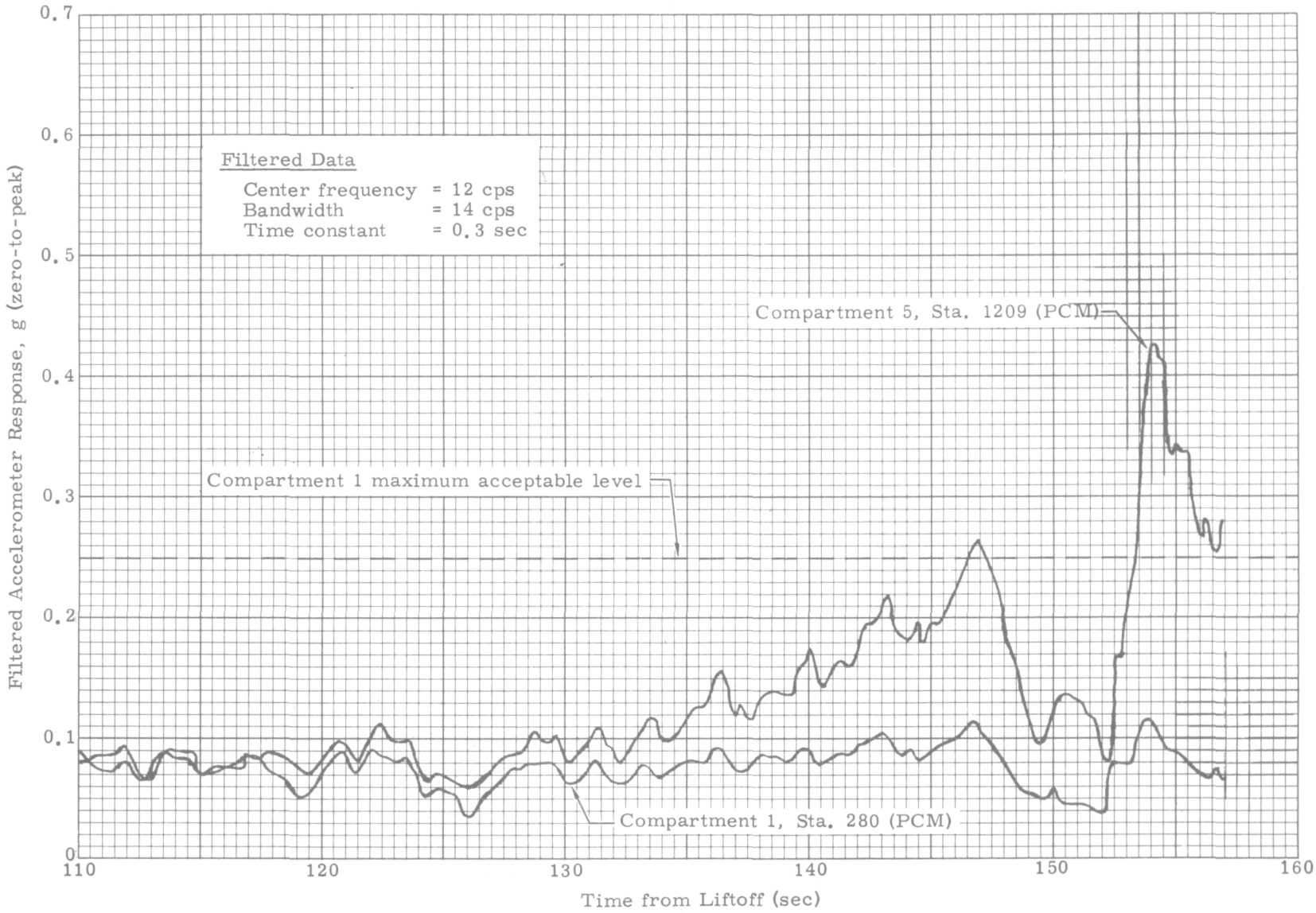


Fig. XII-14. Longitudinal Oscillations

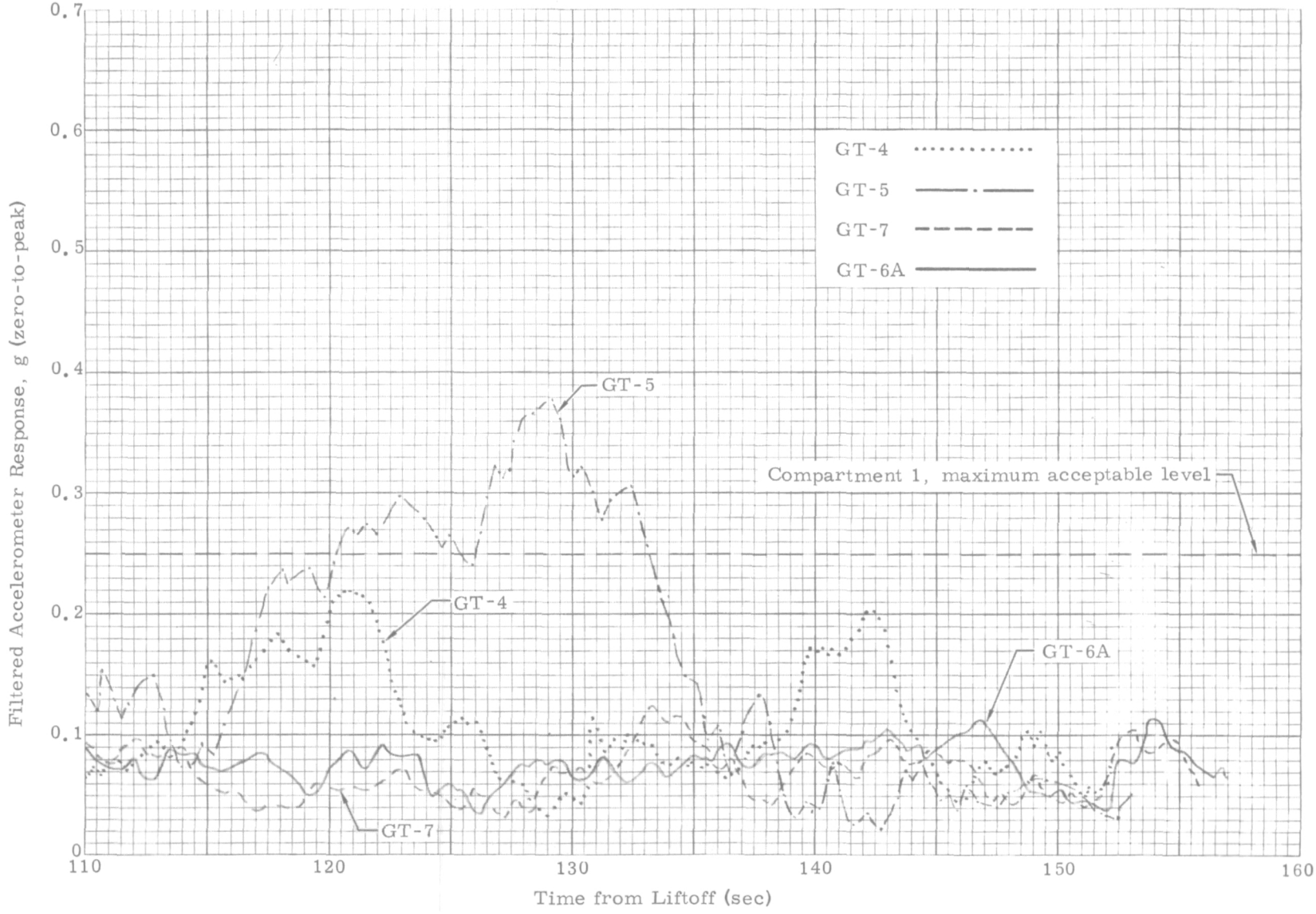


Fig. XII-15. Longitudinal Oscillations, Compartment 1 (Station 280)

XIII. AGE AND FACILITIES

A. MECHANICAL AGE

1. Precount Operations

The mechanical AGE utilized prior to countdown is primarily for transport and erection of Stages I and II. Both stages of GLV-6 were airlifted successfully to Cape Kennedy by B-377PG aircraft. During erection, all equipment functioned as designed.

2. Launch Attempt (12 December 1965)

The countdown proceeded normally through T-0 and engine ignition, but was followed by an automatic shutdown via holdfire C-4. This was caused by premature separation of umbilical 3D1M, one of two pad disconnects which provide a liftoff signal to start the TARS programmer.

As a result of subsequent failure analyses, vibration tests and pull tests, the following corrective action was incorporated into pad disconnect umbilicals 3D1M and 3D2M:

(1) The fairing covering the airborne half of the disconnect was cut back for improved visual inspection.

(2) An index stripe was added to ensure that proper mating of ground and airborne halves is made.

(3) Aluminum lock wire to secure the ground half to airborne half was added to prevent a premature disconnect.

3. Launch Operation (15 December 1965)

Analysis of magnetic tape recordings of functions carried through the umbilicals and inspection of films confirm that all launch vehicle electrical umbilicals separated cleanly in the planned sequence in 0.853 second, as indicated in Table XIII-1.

TABLE XIII-1

Electrical Umbilical Disconnect Sequence

Umbilical Designation	Time of Disconnect (GMT)	Δt (sec)
3D1M/3D2M	1337:26.464	0
3D1E	1337:26.692	0.228

TABLE XIII-1 (continued)

Umbilical Designation	Time of Disconnect (GMT)	Δt (sec)
3D2E	1337:26.878	0.414
3B1E	1337:27.123	0.659
2B1E	1337:27.287	0.823
2B2E	1337:27.317	0.853

B. MASTER OPERATIONS CONTROL SET (MOCS)

1. Launch Attempt (12 December 1965)

Analysis of the MOCS automatic sequence records indicated that the shutdown was initiated by holdfire C-4 (HF-C4, programmer reset check) at T_0 + 1.2 seconds. The shutdown occurred due to a premature disconnect of umbilical pad disconnect 3D1M which started the TARS programmer prior to true liftoff. The holdfire and shutdown circuitry operated properly as designed for this occurrence.

All other MOCS functions prior to shutdown occurred in their proper sequence.

MOCS T_0 occurred at 1454:03.20 GMT, followed by HF-C4 no-go at 1454:04.40 GMT.

2. Launch Operation (15 December 1965)

Analysis of the MOCS automatic sequence records shows that all functions were performed properly. The automatic sequence was picked up at T-35 minutes and proceeded on schedule to the programmed 25-minute hold at T-3 minutes after which the count was resumed and continued through a successful liftoff. MOCS T_0 occurred at 1337:23.13 GMT followed by TCPS "make" 1.04 seconds later. The launch operation was completed in 3.04 seconds.

The recorders were changed to high speed mode at T-2 minutes. During the automatic portion of the count, the operation of the sequencer has been compared to the real-time trace and patch list. All traces were checked for times of occurrence and found to be correct and consistent with the planned operation of the sequencer.

C. ELECTRICAL AGE

The launch and checkout equipment operated satisfactorily during GT-6A prelaunch and launch operations. No problems were encountered.

1. Power Distribution Control Set--PDCS (CP 2800)

The PDCS equipment functioned properly throughout prelaunch and launch operations. The APS, IPS, 25-vdc and inverter monitors did not indicate a hold or shutdown condition, thereby verifying satisfactory operation of the transformer/rectifiers and associated equipment before, during and after power transfer. The interface cabling and umbilicals also functioned satisfactorily.

2. Flight Control System Test Set--FCSTS (CP 2600)

With the exception of the Stage I pitch actuator (4_1), the hydraulic system performed satisfactorily during the launch countdown. At approximately T-21 seconds, the 4_1 position transducer indicated about a 0.8-degree step from null (extend), increasing to 1.7 degrees at Stage I engine ignition. The shutdown null light on the launch monitor and test sequence control panel of the flight control test set went red; however, the actuator position null light remained green. Following Stage I engine ignition transients, the 4_1 position transducer output settled to near null (about 0.2 degree of indicated displacement--extend) and remained at this position through liftoff and umbilical disconnect. The MOCS, properly, did not generate a shutdown.

The flight controls test set actuator position null light indicator is the summation of the four Stage I actuator null switch positions, and the shutdown null light indicator is the summation of the four Stage I position transducer outputs. The MOCS evaluates these two parameters for a shutdown within a specified time period after Stage I ignition. The MOCS generates an automatic shutdown only if both of the following conditions are satisfied:

(1) Any one of the four Stage I actuators is off its null land

(2) Any one of the four Stage I pressure transducers indicates more than 0.7 degree displacement from the null position.

In this case, even if the 4_1 position transducer output displacement had not returned to near null after Stage I engine ignition, the MOCS would not have generated a shutdown.

An investigation of the 4_1 actuator anomaly is in process. A review of the data showed the following:

(1) The flight control test set recorder data and launch vehicle telemetry data agreed.

(2) The flight controls test set circuit involved has been sent to Martin-Baltimore and is undergoing special testing, including piece part analysis in the ASFTS facility.

(3) A circuit analysis and piece part analysis of similar position transducers are also being performed.

D. FACILITIES

All facility items functioned properly throughout the GT-6A countdown and launch.

1. Pad Damage

Damage to AGE and facility items caused by engine blast and heat was minor. All damaged components will be refurbished to their original configuration. The most significant damaged items follow:

Deck Area

The flame shield attached to the inside of the thrust mount ring (east side) was damaged by the engine blast. The damage was confined to one corner of the shield.

Complete Vehicle Erector (CVE)

(1) Personnel elevator rail on east side was broken at the bottom.

(2) Over-speed governor, tension weight support, on the personnel elevator was damaged.

(3) Spacecraft elevator traveling cable duct cover was damaged at the lower end.

(4) Ground strap on the east pivot point was damaged.

(5) Weather curtains were damaged at the 9-foot 8-inch level on the south, west, and northeast sides. At the 15-foot 6-inch level, the curtain on the east side was damaged.

(6) Electrical damage consisted of the following:

(a) The communication J-box conduit on the spacecraft elevator ramp handrail was torn loose.

(b) A light fixture on the northeast corner at the 26-foot 7-inch level was torn loose.

Complete Vehicle Umbilical Tower (CVUT)

(1) One boom cover stiffener angle was blown off boom No. 1.

(2) Elevator cable guard screens attached to the tower were damaged on CVUT levels 1, 2, and 4.

Second Stage Umbilical Tower (SSUT)

(1) Cable duct cover was blown off at level No. 1.

(2) Spacecraft elevator cable guard screen was bent inward at level No. 1.

(3) All lights and conduits were damaged on top of the tower in back of Panel S32L.

XIV. RELIABILITY

Based on countdown experience through GLV-7, the average number of holds per countdown ($\bar{h}$) was calculated to be 0.125, i.e., one hold per eight countdowns. The probability of GLV-6 (12 December 1965 attempt) completing the countdown without a hold was predicted to be

$$P_{C/D}\,(\bar{h} = 0.125) = 0.88$$

Including the GLV-6 (15 December 1965 launch) countdown, the average number of holds per countdown ($\bar{h}$) is shown in Table XIV-1 to be 0.1, i.e., one hold in ten countdowns. The probability of GLV-8 completing countdown without a hold is predicted to be

$$P_{C/D}\,(\bar{h} = 0.1) = 0.90$$

$\bar{h}$ is based on the countdown period from T-240 minutes to T-0, except for the GLV-5 attempt which was scrubbed at T-10 minutes due to weather but which counted as one countdown without a hold. Spacecraft holds and SCF tests were not counted.

The GLV-6 (12 December 1965 attempt) was scrubbed due to a shutdown that occurred after T-0. The shutdown was caused by an umbilical prematurely disconnecting (a dust cap was later found in the oxidizer gas generator line). Since the shutdown occurred after T-0, GLV-6 (12 December 1965 attempt) was used in Table XIV-1 as one countdown without a hold.

Countdown experience for all GLVs is included in Table XIV-1.

TABLE XIV-1

Countdown Experience Including GT-7 Launch

Vehicle No.	No. of Countdowns	No. of Holds*	Remarks
GLV-1	1	0	
GLV-2 (attempt)	1	0	3 S/C holds. Tandem actuator failed after T-0
GLV-2	1	0	1 S/C hold
GLV-3	1	0	1 hold--not Martin responsibility (oxidizer leak in Stage I engine transducer)
GLV-4	1	1	Erector stuck during lowering
GLV-5 (attempt)	1	0	1 S/C hold. Incomplete countdown--scrubbed at T-10 min due to weather
GLV-5	1	0	
GLV-6 (attempt on 25 Oct 65)	0	0	Incomplete countdown--scrubbed at T-42 min due to Agena failure
GLV-7	1	0	
GLV-6 (attempt on 12 Dec 65)	1	0	Umbilical failed (prematurely disconnected) after T-0, and dust cap in oxidizer gas generator line
GLV-6 (launch 15 Dec 65)	1	0	
Total	10	1	

*Based on Martin holds only.

XV. RANGE DATA

A. LAUNCH ATTEMPT DATA AND FILM DISTRIBUTION

1. Launch Attempt Data

The following GLV-6 launch attempt (12 December 1965) data were supplied to Martin-Baltimore within 24 hours after the attempt:

(1) Range supplied quick-look data

  (a) Telemetry magnetic tapes

    (i) Tel II, post-detected, PCM/FM

    (ii) Tel II, formatted

(2) Martin data

  (a) Landline records (events, Bristol, Multipoint, Sanborn) with associated calibrations

  (b) BLH tabulation (fueling and defueling)

  (c) Fueling and oxidizer loading and detanking records

  (d) CP 2600 records (2612, 2650 and 2660)

  (e) Sequencer records and code sheets

  (f) Dub of Complex 19 landline tape

(3) Range supplied data

  (a) Surface weather observations

  (b) Upper triple theodolite

  (c) Weather tower 700/701.

2. Film Distribution

The following launch attempt film was supplied as a follow-on request to be made available as soon as possible.

1.2-14 16mm, explosive bolts and first motion

1.2-15 16mm, explosive bolts and first motion

1.2-16 16mm, east launch ring engine

1.2-17 16mm, west launch ring engine

1.2-18 16mm, north launch ring engine

1.2-19 16mm, south launch ring engine

1.2-20 16mm, 3D1E and 3D2E umbilical plugs

B. LAUNCH DATA DISTRIBUTION

1. Quick-Look Range Data

All available quick-look data from the 15 December 1965 launch were supplied by ETR to Martin-Baltimore as shown in Table XV-1.

The PCM serial tape was of good quality and exhibited minor drop-outs. The quick-look formatted tape was of good quality except for bad time words from LO + 294 to LO + 297 seconds. The final formatted magnetic tape was of excellent quality and contained no redundancies. Except for approximately 300 milliseconds of transmission blackout during booster staging, the Tel II formatted tape showed that there were only five bad data words and one bad frame from LO - 10 seconds to LO + 420 seconds.

TABLE XV-1
Range Supplied Quick-Look Data

Description	Time Requested	Time Received (ETR)	Time Received (Baltimore)
Telemetry magnetic tapes:			
Tel II, Post-detection PCM/FM (1 roll)	T + 1 hr	T + 1 hr	T + 10 hr
Station 1 formatted (3 rolls)	T + 4 hr	T + 4 hr	T + 10 hr

2. Martin Data

Test data and records acquired and generated by Martin at Cape Kennedy were received in Baltimore within two days after launch. These data consisted of the following items:

(1) One set of quick-look records from RCA tape

(2) High speed records of engine parameters

(3) Landline records (events, Bristol, Multipoint and Sanborn) with associated calibrations

(4) BLH tabulation

(5) CP 2600 records (2650, 2660 and events)

(6) Sequencer records with code sheets

(7) Summary of flight events

(8) Dub of Complex 19 landline magnetic tape

(9) Fuel and oxidizer loading records.

3. Range Data

All data supplied by the ETR are summarized in Table XV-2. The time requested for delivery to Martin-Canaveral (Ref. 6555th ATW Form 1-116, dated 7 December 1965) and the time received at Baltimore are shown in this table.

TABLE XV-2

Range-Supplied Data

OD Item No.	Description	Time Requested (Canaveral)	Time Received (Baltimore)
8	Position, velocity and acceleration, and special parameters	4 CD	12 CD
19	Position, velocity and acceleration, MISTRAM I	5 CD	7 CD
20	Position, velocity and acceleration, MISTRAM I and MISTRAM II	11 WD	16 WD
5	Attitude, camera	3 CD	7 CD
4.9/29.9	MISTRAM function recordings	3 WD	6 WD

TABLE XV-2 (continued)

OD Item No.	Description	Time Requested (Canaveral)	Time Received (Baltimore)
26	Best estimate of trajectory	15 WD	18 WD
1.5.2	Serial PCM, post-detection magnetic tape, FR 600: Quick-look Final	1 hr 24 hr	1 hr *
b.	PCM formatted, quick-look	4 hr	(See item c final)
c.	PCM formatted final	9 hr	3 WD
e.	PCM formatted	1 CD	7 CD
3.5-2	Serial PCM, post-detection magnetic tape	3 CD	*
1.5-7	Signal strength (center frequency) recordings	24 hr	*
1.5-53	Signal strength (center frequency) recordings	24 hr	*
3.5-5	Signal strength (center frequency) recordings	3 CD	1 CD
7.5-3	Signal strength (center frequency) recordings	3 CD	*
55	Tracking system comparisons, MOD III/MISTRAM I	7 CD	12 CD
56	Comparisons involving adjusted trajectory	20 CD	26 CD
1.5-9	Oscillograph records, near real-time	1 CD	1 CD

TABLE XV-2 (continued)

OD Item No.	Description	Time Requested (Canaveral)	Time Received (Baltimore)
1.11-4	Instrumentation data logs	3 CD	*
1.18	Range safety plot charts	4 hr	4 hr
4.7.3.1	Real-time computer facility metric data	24 hr	*
1.11-1	Command control function records	3 WD	*
1.11-5	Command control function records	3 WD	*
1.11-6	Command control function records	3 WD	*
3.11-1	Command control function records	5 WD	*
3.11-3	Command control function records	5 WD	*
7.11-1	Command control function records	5 WD	*
7.11-3	Command control function records	5 WD	*
2.4.1	Preliminary test report	2 hr	16 WD
5.4.1	Propellant analysis report	2 WD	*
60(e)	Weather surface observations	1 WD	6 hr
60(g)	Weather upper theodolite, triple	1 WD	24 hr

TABLE XV-2 (continued)

OD Item No.	Description	Time Requested (Canaveral)	Time Received (Baltimore)
60(k)	Weather upper Rawinsonde	1 WD	6 hr
60(a)	Weather tower 700/702	1 WD	6 hr
34	Trendplots, QLAP, Part I	32 hr	7 CD
36	Transient plots, Part I	32 hr	7 CD
1.5-9	Bilevel oscillographs, Stations 1 and 3	1 CD	*
3.11-25	Command control antenna position (GBI)	3 CD	*
	Error analysis report	60 CD	*

* Data not received by 18 January 1966

CD = Calendar days
WD = Working days

4. Agency/Contractor Supplied Data

Table XV-3 presents data received from associated contractors and NASA-MSC.

TABLE XV-3

Agency/Contractor Supplied Data

Description	Supplier	Received (Baltimore)
Mod III-G, AMRO guided missile control facility	GE, ETR	2 CD
Mod III-G, radio guidance system	GE, Syracuse	4 CD
S/C re-entry TLM recordings	NASA	20 CD

C. LAUNCH FILM COVERAGE

Photographic conditions at Cape Kennedy preceding and during the GT-6A launch (15 December 1965) were excellent, and motion picture coverage was very good. Table XV-4 contains a listing of the films obtained from the fixed cameras and the tracking cameras.

The 70-mm tracking films (Items 1.2-40, 1.2-41 and 1.2-42) were reviewed for information pertaining to the booster staging event. Inspection of these films shows that the normal breakup of the first stage transportation section occurred after Stage II had separated from Stage I.

TABLE XV-4

Film Coverage and Disposition

OD Item No.	Description	Footage (ft)	Time Received (Baltimore)
1.2-9 16 mm fixed	Missile centered, 50-ft tower 19-1	180	8 WD
1.2-12 16 mm fixed	Spacecraft centered, 50-ft tower 19-7A	81	8 WD
1.2-13 16 mm fixed	Spacecraft centered, 50-ft tower 19-2	80	3 WD
1.2015 16 mm fixed	Explosive bolts and first motion, 50-ft tower 19-7A	110	4 WD
1.2-16 16 mm fixed	East launch ring, engine	170	1 WD
1.2-17 16 mm fixed	West launch ring, engine	130	4 WD
1.2-18 16 mm fixed	North launch ring, engine	200	8 WD
1.2-19 16 mm fixed	South launch ring, engine	155	8 WD
1.2-20 16 mm fixed	3D1E and 3D2E umbilical plugs, umbilical tower, first level	140	3 WD
1.2-21 16 mm fixed	2DFVT umbilical plug, umbilical tower, second level	146	4 WD
1.2-22 16 mm fixed	1DOVT umbilical plug, umbilical tower, fourth level	200	4 WD
1.2-23 16 mm fixed	3B1E umbilical plug, umbilical tower, fifth level	150	3 WD
1.2-24 16 mm fixed	2B1E and associated umbilical plugs, umbilical tower, sixth level	140	8 WD
1.2-25 16 mm fixed	Umbilical Boom 3, "J" bars and lanyards, umbilical tower, sixth level	210	8 WD
1.2-26 16 mm fixed	Spacecraft upper umbilical plug, first level	80	8 WD
1.2-27 16 mm fixed	Spacecraft lower umbilical plug, second level	140	4 WD
1.2-31 16 mm tracking	LO to LOV pad emergency, spacecraft centered, U191L122	337	8 WD
1.2-32 16 mm tracking	LO to LOV, spacecraft centered, U176L46	350	8 WD
1.2-33 16 mm tracking	LO to LOV, spacecraft centered, U122L29	385	4 WD

TABLE XV-4 (continued)

OD Item No.	Description	Footage (ft)	Time Received (Baltimore)
1.2-34 16 mm tracking	LO to LOV, missile centered, U102R6	310	4 WD
1.2-35 16 mm tracking	LO to LOV, missile centered, U58R41	375	8 WD
1.2-36 16 mm tracking	First acquisition to LOV, missile centered, U112L24	250	4 WD
1.2-37 16 mm tracking	First acquisition to LOV, missile centered, U247L116	360	8 WD
1.2-38 16 mm tracking	First acquisition to LOV, missile centered, U29R46	377	8 WD
1.2-39 16 mm tracking	First acquisition to LOV, missile centered through staging (IGOR) IGPA	180	8 WD
1.2-39 35 mm tracking	First acquisition to LOV, missile centered through staging (IGOR) IGPA	450	8 WD
1.2-40 16 mm tracking	First acquisition to LOV, missile centered through staging, UCS 12	132	8 WD
1.2-40 70 mm tracking	First acquisition to LOV, missile centered through staging, UCS 12	1006	8 WD
1.2-41 16 mm tracking	First acquisition to LOV, missile centered through staging (ROTI) RTCB	130	8 WD
1.2-41 70 mm tracking	First acquisition to LOV, missile centered through staging (ROTI) RTCB	990	8 WD
1.2-42 16 mm tracking	First acquisition to LOV, missile centered through staging (ROTI) RTMB	125	8 WD
1.2-42 70 mm tracking	First acquisition to LOV, missile centered through staging (ROTI) RTMB	30 & 966	8 WD

WD = Working days
CD = Calendar days

XVI. PRELAUNCH AND COUNTDOWN OPERATIONS

A. PRELAUNCH

1. Simulated Flight Test

The GT-6A simulated flight test (SFT) was performed successfully on 8 December 1965 in accordance with the Martin test procedure (Ref. 8). The flight crew was in the spacecraft during the primary run, and the backup crew was aboard during the secondary run for spacecraft monitoring and training.

The countdown for the secondary run was started at T-45 minutes (1302 EST) and was successfully completed at T+6 minutes (1353 EST). The primary run was started at T-3 minutes (1448 EST) and was successfully completed at T+6 minutes (1457 EST).

2. Launch Attempt Precountdown Activities

Final power application to the launch vehicle occurred at 0800 EST, 11 December 1965. The precountdown tests were started at 1200 EST with the range sequencer at T-770 minutes. The tests were successfully performed and the range sequencer was secured at T-530 minutes (1630 EST).

Propellant loading started on schedule at 1800 EST and was completed in record time at 2110 EST. The range sequencer was restarted at T-530 minutes (on schedule) at 0104 EST, 12 December 1965.

B. LAUNCH ATTEMPT COUNTDOWN SUMMARY

The countdown was picked up on schedule at 0529 EST on 12 December 1965. The 240-minute countdown was performed in accordance with the Martin test procedure (Ref. 9). The countdown progressed smoothly and astronaut ingress occurred at approximately T-100 minutes. A programmed 25-minute hold was initiated at T-3 minutes (0926 EST). The countdown resumed at T-3 minutes (0951 EST) and progressed through T-0 (0954 EST) engine ignition, followed by an automatic shutdown at T+1.2 seconds because of tail plug 3D1M disengaging prior to liftoff. The propellant tanks were vented immediately, and preparations were made to raise the erector for astronaut egress. The erector was raised at 1128 EST, and the astronauts left the spacecraft at 1133 EST.

After the launch vehicle had been secured, the ground half of Disconnect 3D1M was removed for failure analysis and a replacement disconnect installed and safety wired.

Evaluation of the launch attempt data showed that performance degradation was occurring in S/A 2 prior to engine shutdown. Consequently, the gas generator and bootstrap lines were removed for inspection. Investigation revealed that a plastic dust cap had been inadvertently left in the gas generator assembly.

C. RECYCLE AND PRELAUNCH ACTIVITY

1. Recycle

Recycle started immediately in preparation for a launch scheduled for 0837 EST on 15 December 1965. The recycle was performed in accordance with GLV Test and Checkout Specification 424-1430002, with minor exceptions as noted in Ref. 14.

2. Precountdown Activities

Final power application to the launch vehicle occurred at 1700 EST on 14 December 1965. Oxidizer loading started on schedule at 2012 EST and was completed at 2155 EST. Fuel loading started at 2230 EST and was completed at 2332 EST.

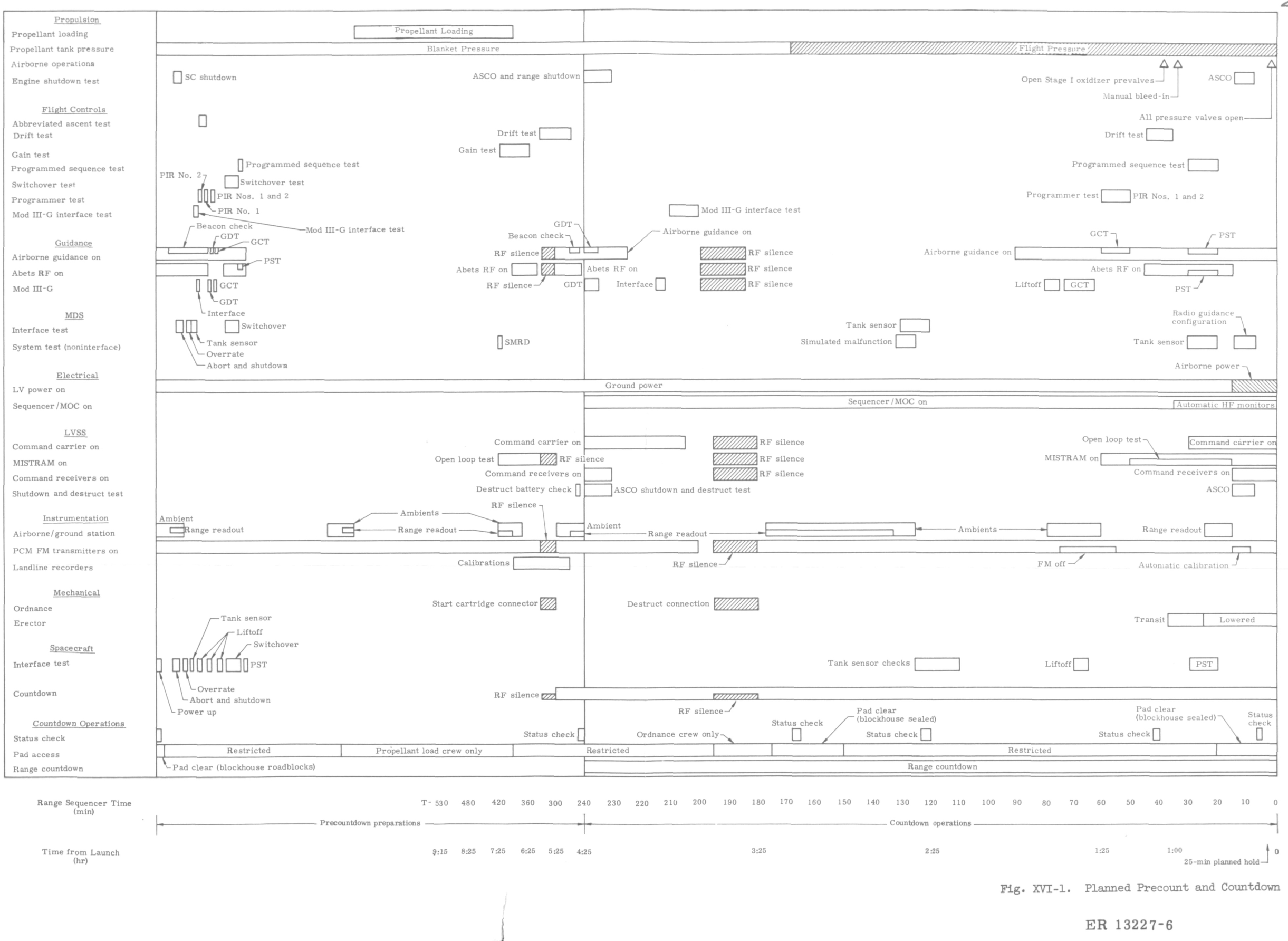
The range sequencer was started on schedule at T-530 minutes at 2322 EST on 14 December 1965. All precountdown tests were completed prior to starting the 240-minute countdown.

D. COUNTDOWN SUMMARY

The launch countdown was picked up on schedule at 0412 EST on 15 December 1965. The 240-minute countdown was performed in accordance with Martin test procedure (Ref. 9). The countdown progressed smoothly and astronaut ingress occurred at approximately T-97 minutes. A programmed 25-minute hold was initiated at T-3 minutes (0809 EST). The countdown resumed at T-3 minutes (0834 EST), and liftoff occurred exactly on schedule at 0837:26 EST.

The countdown schedule is shown in Fig. XVI-1.

XVI-3-1



Range Sequencer Time (min)	T- 530	480	420	360	300	240	230	220	210	200	190	180	170	160	150	140	130	120	110	100	90	80	70	60	50	40	30	20	10	0
	Precountdown preparations						Countdown operations																							
Time from Launch (hr)		9:15	8:25	7:25	6:25	5:25	4:25					3:25					2:25							1:25		1:00			25-min planned hold	0

Fig. XVI-1. Planned Precount and Countdown

ER 13227-6

XVII. CONFIGURATION SUMMARY

A. LAUNCH VEHICLE SYSTEMS DESCRIPTION

The Gemini Launch Vehicle (GLV) is a modified two-stage Titan II intercontinental ballistics missile (ICBM) which has been "man rated" for Gemini usage. The propulsion system in each stage uses hypergolic (self-igniting upon mixture) propellants. Modifications to the basic Titan II vehicle to achieve the "man rated" GLV follow:

(1) Addition of a completely redundant malfunction detection system (MDS).

(2) Replacement of the Titan II inertial guidance system (IGS) with the Mod III-G radio guidance system (RGS).

(3) Addition of a three-axis reference system (TARS) to provide attitude reference and open-loop programming to the autopilot.

(4) Addition of a secondary flight control system (FCS).

(5) Addition of a secondary Stage I hydraulic system.

(6) Addition of the capability of switchover to the secondary guidance, flight control, and hydraulic systems.

(7) Provision of redundancy in electrical sequencing by APS and IPS power.

(8) Provision of an engine shutdown capability from the spacecraft.

(9) Provision of a 120-inch diameter cylindrical skirt forward of the Stage II oxidizer tank for mating the spacecraft to the launch vehicle.

(10) Removal of the retrorockets, vernier rockets and associated equipment.

(11) Addition of fuel line spring-piston accumulators and oxidizer line tuned standpipes for suppression of POGO vibrations.

(12) Capability for redundant Stage II engine shutdown (GLV-3 and up).

Significant GLV-7 changes from the GLV-5 configuration are listed in Table XVII-1.

TABLE XVII-1

GLV-6 Modifications

System	Significant Configuration Changes from GLV-5
Structures	1. Added conduit vacuum test and helium leak test after hydro changed from high pressure to low pressure on Stages I and II 2. Changed location of drain holes at skirt-to-dome weld from between dome chord splices to dome chord splices in all compartments
Propulsion	1. Reduced diameter of oxidizer pressurant back pressure orifice from 0.50 to 0.46 inch.
Flight controls	1. Added 0.5- to 10-second delay timer to change the time of gain change No. 1 from 104.96 to 110.0 seconds
Guidance	None
Hydraulics	1. Added check valve to electric motor pump discharge port
Electrical	None
Malfunction detection	None
Instrumentation	None
Range safety and ordnance	None
AGE	1. Provided tandem flowmeters for each tank 2. Provided redundancy in propellant transfer system

A detailed description of all GLV systems is presented in Martin Engineering Report, "Launch Vehicle No. 4 Flight Evaluation" (Ref. 3).

B. MAJOR COMPONENTS

The two major GT-6A components were as follows:

(1) Spacecraft

(a) Manufacturer: McDonnell Aircraft Corporation

(b) Serial Number: Spacecraft Number 6

(2) Gemini Launch Vehicle

(a) Manufacturer: Martin Company

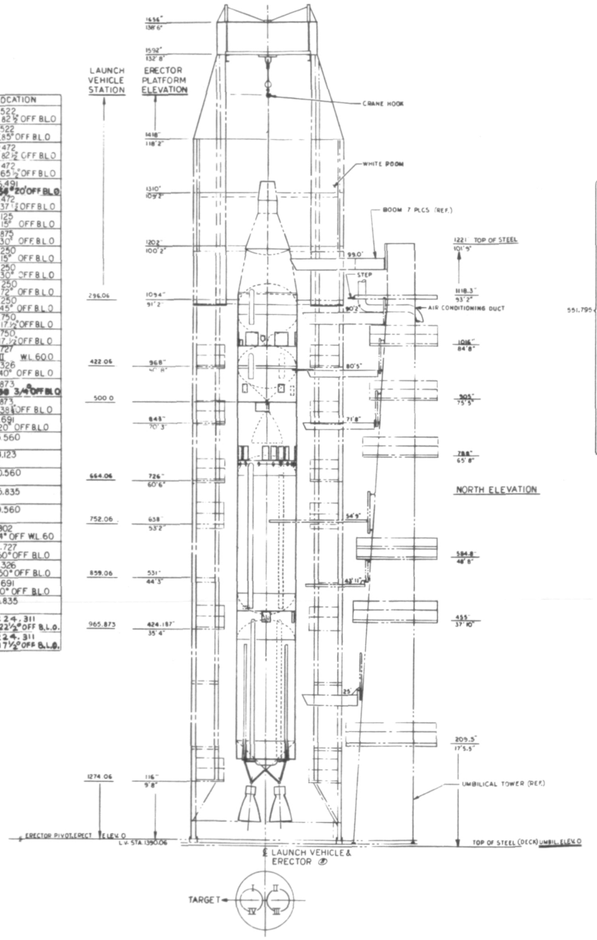
(b) Serial Number: GLV-6

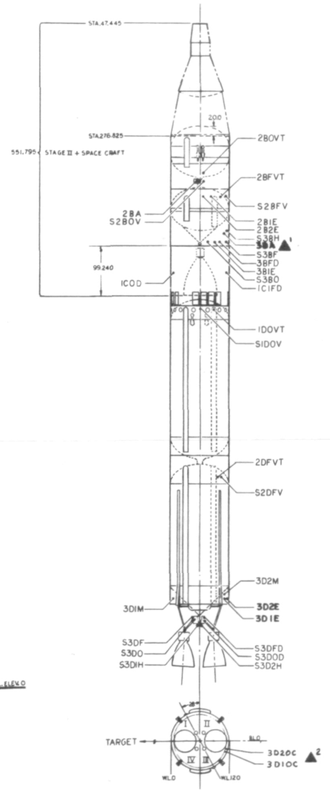
(c) Air Force Serial Number: 62-12561.

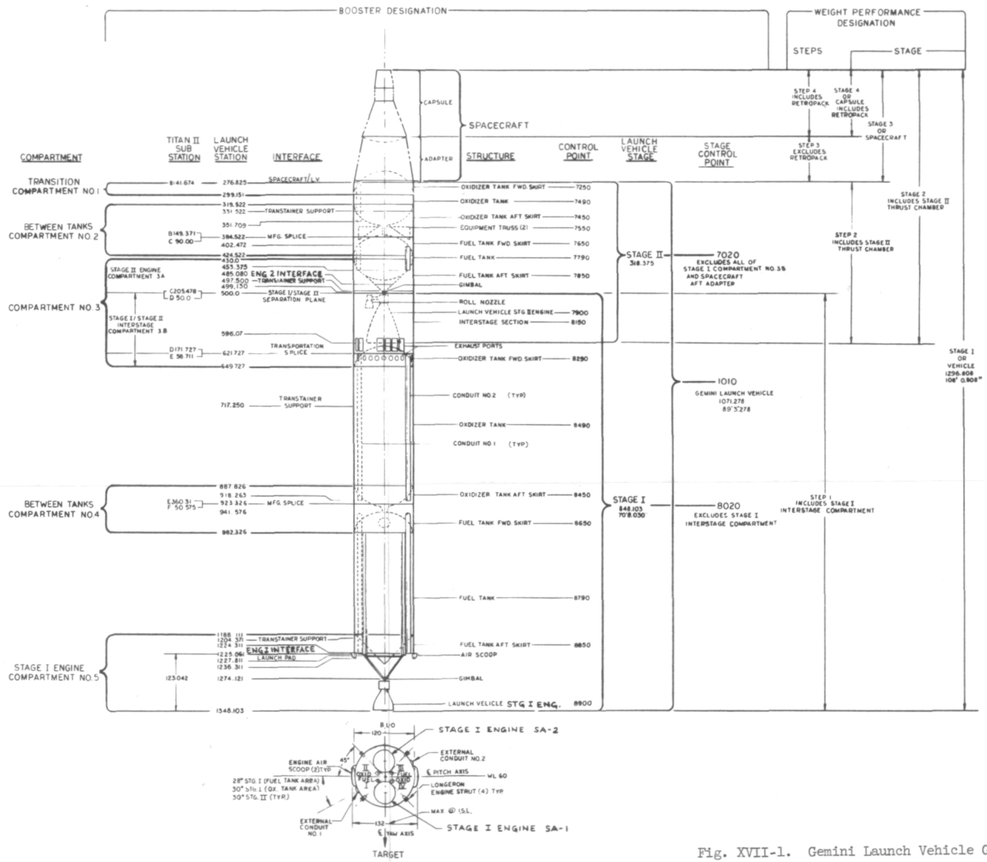
Figure XVII-1 shows the general arrangement of the GLV.

LAUNCH VEHICLE UMBILICAL SYSTEMS INTERFACE

CODE	FUNCTION	SYSTEM	LOCATION
28A	AIR COND.	GUIDANCE AIR COND. STG. II	STA. 371.522 QUAD I, II 82½° OFF BL.O
28OV	OXID. VENT	OXID STG. II	STA. 371.522 QUAD. II 65° OFF BL.O
28E	ELECTRICAL	ELECTRICAL STG. II	STA. 402.472 QUAD. II 82½ OFF BL.O
28IE	ELECTRICAL	ELECTRICAL STG. II	STA. 402.472 QUAD. II 65½° OFF BL.O
28FVT	FUEL VENT TOPPING	FUEL STAGE II	STA. 406.481 QUAD I, II 50° OFF BL.O
28FV	FUEL VENT	FUEL STG II	STA. 404.472 QUAD. II 37½° OFF BL.O
38H	HYDRAULIC	HYDRAULIC STG II	STA. 470.125 QUAD II 15° OFF BL.O
▲ 38A	AIR CONDITIONING	ENGINE AREA STG II	STA. 474.675 QUAD II 30° OFF BL.O
38F	FUEL FILL AND DRAIN	FUEL STG II	STA. 492.250 QUAD II 15° OFF BL.O
38FD	FUEL PRESS. SEQ. VALVE DRAIN	FUEL STG II	STA. 492.250 QUAD II 30° OFF BL.O
38O	OXID. FILL AND DRAIN	OXID STG II	STA. 492.250 QUAD II 45° OFF BL.O
38IE	ELECTRICAL	ELECTRICAL STG II	STA. 492.250 QUAD II 45° OFF BL.O
ICIFD	TUBO-PUMP FUEL DRAIN	FUEL STG I	STA. 505.750 QUAD II, III 2½° OFF BL.O
ICOD	TUBO-PUMP OXID. DRAIN	OXID STG I	STA. 505.750 QUAD I, II, III 2½° OFF BL.O
SIDOV	OXID. VENT	OXID. STG I	STA. 530.727 QUAD I & II WL 60.0
S2DFV	FUEL VENT	FUEL STG I	STA. 963.326 QUAD II 40° OFF BL.O
3DIM	MALFUNCTION DETECTION SYS.	MALFUNCTION STG. I	STA. 214.873 QUAD I 30° 3.4° OFF BL.O
3D2M	MALFUNCTION DETECTION SYS.	MALFUNCTION STG. I	STA. 214.873 QUAD III 30° OFF BL.O
3DIE	ELECTRICAL	ELECTRICAL STG. I	STA. 216.691 QUAD II 20° OFF BL.O
S3DF	FUEL FILL AND DRAIN	FUEL STG. I	STA. 243.560 QUAD. I
S3DFD	FUEL DRAIN - PRE-VALVE	FUEL STG I	STA. 244.123 QUAD III
S3DOD	OXID. DRAIN - PRE-VALVE	OXID. STG. I	STA. 250.560 QUAD II
S3DIH	HYDRAULIC SUPPLY AND RETURN	HYDRAULIC STG. I PRIMARY	STA. 255.835 QUAD I
S3DO	OXID. FILL AND DRAIN	OXID. STG I	STA. 249.560 QUAD IV
2BOVT	OXID. VENT TOPPING	OXID. STG. II	STA. 351.802 QUAD II 4° OFF WL 60
IDOVT	OXID. VENT TOPPING	OXID. STG. I	STA. 630.727 QUAD II 60° OFF BL.O
2DFVT	FUEL VENT TOPPING	FUEL STAGE I	STA. 963.326 QUAD II 50° OFF BL.O
302E	ELECTRICAL	ELECTRICAL STG. I	STA. 216.691 QUAD II 10° OFF BL.O
S302H	HYDRAULIC SUPPLY AND RETURN	HYDRAULIC STG. I SECONDARY	STA. 255.835 QUAD II
▲ 3D10C	OXID. REMOTE CHARGING SYSTEM	OXID. STG. I	STA. 12.2-4.311 QUAD III 22½° OFF B.L.O.
▲ 3D20C	OXID. REMOTE CHARGING SYSTEM	OXID. STG. I	STA. 12.2-4.311 QUAD III 17½° OFF B.L.O.







4. ▲2 GLV 4 & UP
3. ▲ GLV 1 THRU 4 ONLY
2. THIS DRAWING RELEASES NO PARTS REF. ONLY.
1. NEW GEMINI LAUNCH VEHICLE DRAWING

NOTE:

Fig. XVII-1. Gemini Launch Vehicle General Arrangement

XVIII. REFERENCES

1. "Launch Vehicle No. 7 Flight Evaluation." Engineering Report 13227-7, Martin Company, Baltimore, Maryland, January 1966. Confidential

2. "Launch Vehicle No. 5 Flight Evaluation." Engineering Report 13227-5, Martin Company, Baltimore, Maryland, October 1965. Confidential

3. "Launch Vehicle No. 4 Flight Evaluation." Engineering Report 13227-4, Martin Company, Baltimore, Maryland, July 1965. Confidential

4. "Launch Vehicle No. 3 Flight Evaluation." Engineering Report 13227-3, Martin Company, Baltimore, Maryland, May 1965. Confidential

5. "Launch Vehicle No. 2 Flight Evaluation." Engineering Report 13227-2, Martin Company, Baltimore, Maryland, March 1965. Confidential

6. "Launch Vehicle No. 2 Launch Attempt Evaluation." Engineering Report 13227-2X, Martin Company, Baltimore, Maryland, January 1965. Confidential

7. "Launch Vehicle No. 1 Flight Evaluation." Engineering Report 13227-1, Martin Company, Baltimore, Maryland, May 1964. Confidential

8. "Martin-Canaveral Test Procedure." 424-876-ETR, Revision F.

9. "Martin-Canaveral Test Procedure." 424-875-ETR, Revision N.

10. "GT-6A Pre-flight Report." LV-326-6, Martin Company, Baltimore, Maryland, November 1965. Confidential

11. "Gemini Launch Vehicle Performance Specification." MB-1046, SCN-10, Martin Company, Baltimore, Maryland, 20 December 1965. Confidential

12. Letter from SSD to MMB, dated 19 October 1965, subject: "Transmittal of RGS Dispersions for GT-6."

13. "Flight Weight Coordination Report, Post-Flight Weight, GT-6." LV-165-6B, Martin Company, Baltimore, Maryland, 19 January 1966.

14. Martin letter MG-2012 to SSD, dated 14 December 1965, subject: "Test and Checkout Specification Waiver."

15. "Subsystem Engineering Analysis YLR 87-AJ-5 and YLR 91-AJ-5 Rocket Engines." AGC 521-3.15 Q-15, Aerojet-General Corporation, Sacramento, California, 22 July 1964. Confidential

16. "Stage I Base Environments Analysis." LV-163, Martin Company, Baltimore, Maryland, January 1963.

17. "Master Measurements List." LV-220, Revision N, Martin Company, Baltimore, Maryland, 24 January 1966.

18. "Aerojet Engine Test Directive." 2.1-3.3E, Aerojet-General Corporation, Sacramento, California, 22 July 1964.

APPENDIX A

SUMMARY OF GEMINI LAUNCHES

CONFIDENTIAL A2-2

Summary of Gemini Launches

Mission	Launch Date and Time (hr EST)	Launch Vehicle Payload (lb)	Burning Time Stage I (sec)	Burning Time Stage II (sec)	Inertial Velocity (fps) BECO	Inertial Velocity (fps) SECO	Inertial Velocity (fps) SECO + 20 Sec	Altitude (ft) BECO	Altitude (ft) SECO	Altitude (ft) SECO + 20 Sec	Inertial Flight Path Angle (deg) BECO	Inertial Flight Path Angle (deg) SECO	Inertial Flight Path Angle (deg) SECO + 20 Sec	Time in Orbit ③ (hr) Stage II	Time in Orbit ③ (hr) Spacecraft	Orbit (naut mi) Apogee	Orbit (naut mi) Perigee	Launch Evaluation Report Number
GT-1	4-8-64 1100	7029 ①	157.5	185.3	9,752	25,679	25,786	208,262	531,500	528,184	20.00	0.0	-0.03	95.2 ① (64 orbits)	95.2 ① (64 orbits)	173	86.6	ER 13227-1
GT-2	1-19-65 0904	6890 ②	155.1	180.4	9,916	25,611	25,738	229,743	546,960	526,380	26.219	-2.4523	-2.3431	N/A ②	N/A ②	N/A ②	N/A ②	ER 13227-2X ④ ER 13227-2
GT-3	3-23-65 0924	7112	155.8	181.3	9,981	25,587	25,688	224,777	531,477	532,338	21.79	0.0	0.0323	18 (13 orbits)	4.6 (3 orbits)	121	87	ER 13227-3
GT-4	6-3-65 1016	7868	155.7	181.3	9,844	25,670	25,745	214,775	531,522	532,886	18.66	-0.0235	0.059	47.7 (34 orbits)	97.7 (66 orbits)	152.3	87	ER 13227-4
GT-5	8-21-65 0900	7947	156.8	179.7	9,848	25,713	25,806	215,607	531,276	531,118	19.90	-0.0279	-0.0129	72 (51 orbits)	190.9 (127.9 orbits)	189	87	ER 13227-5
GT-7	12-4-65 1430	8085	159.1	181.4	10,049	25,735	25,789	207,088	529,583	529,738	18.66	0.0500	0.0285	66 (46.6 orbits)	330.6 (219.8 orbits)	177.1	87	ER 13227-7
GT-6A	12-15-65 0837	7821	160.4	181.6	9,992	25,634	25,728	202,186	529,891	530,201	17.94	0.08	-0.054	31 (21 orbits)	25.8 (17.1 orbits)	140.4	87	ER 13227-6

① Spacecraft and Stage II inserted into orbit as a unit.

② Suborbital mission (spacecraft impact 2125 miles downrange).

③ Inertial orbit.

④ Launch attempt report.

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